No. 12885

United States Court of Appeals

for the Ninth Circuit.

CONSOLIDATED VULTEE AIRCRAFT CORPORATION and AMERICAN AIR LINES, INC.,

Appellants,

vs.

MAURICE A. GARBELL, INC., and GARBELL RESEARCH FOUNDATION,

Appellees.

SUPPLEMENTAL Transcript of Record

Volume V

Book of Exhibits

(Pages 1007 to 1137)

Appeal from the United States District Court for the Southern District of California, Central Division.

Phillips & Van Orden Co., 870 Brannan Street, San Francisco, Calif.

INDEX

[Clerk's Note: When deemed likely to be of an important nature, errors or doubtful matters appearing in the original certified record are printed literally in italic; and, likewise, cancelled matter appearing in the original certified record is printed and cancelled herein accordingly. When possible, an omission from the text is indicated by printing in italic the two words between which the omission seems to occur.]

DEFENDANTS' EXHIBIT A

Page 1 of 60

Consolidated Aircraft Corporation
San Diego, California

Model..........Airplane Report No. ZA-101

Report on Airfoil Selection for the Revised Two-Engine Tailless Design
ZA-101

February 25, 1944

By Abraham Firel
M. A. Garbell
M. Rogers

Approved: [Illegible]

Page 2 of 60

Consolidated Vultee Aircraft Corporation
San Diego Division

Model..........Airplane Report No. ZA-101

Foreword

This report summarizes the concepts and procedures used in the selection of airfoils for the revised two-engine tailless design.

Defendants' Exhibit A—(Continued)

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Consolidated Vultee Aircraft Corporation
San Diego Division

Model..........Airplane Report No. ZA-101

Table of Contents

Defendants' Exhibit A—(Continued)

Consolidated Vultee Aircraft Corporation
San Diego Division

Model..........Airplane Report No. ZA-101

Summary

New airfoils were selected for the revised two-engine tailless design to satisfy the following design requirements:

1. Good stalling characteristics with elevators neutral and deflected upward;

2. More conservative chordwise load distribution to retard the premature separation observed on the original airfoils;

3. Higher maximum lift with flaps retracted;

4. Slightly greater positive pitching moment coefficient at zero lift to raise the trim lift coefficient.

The new airfoils selected are defined by the following parameters:

Airfoil Family	Design Lift Coefficient C_{li}	Maximum Thickness % Chord	Chordwise Load Parameters a	b	Spanwise Location
63,4	.1	22	.1	.59	Root
63,4	.3	18	.1	.59	.48 Semi-Span
63,4	.5	16	.1	.59	Tip

The three airfoils are to be placed in a tapered wing of aspect ratio 12, taper ratio 4:1, leading-edge sweepback 11°-24′, with one degree aerodynamic washout at .48 semi-span and at the tip, referred to the root chord.

Defendants' Exhibit A—(Continued)

Data used in the selection of these airfoils are given in the text of the report and in the appedices. The geometric characteristics of the airfols and wing may be obtained from the various tables and charts.

Consolidated Vultee Aircraft Corporation
San Diego Division

Model..........Airplane Report No. ZA-101

Airfoil Selection & Wing Design

Structural and balance considerations, rather than the aerodynamicist's judgment, often determine the design of conventional wings. However, as stability and good stalling characteristics were to be the major criteria in the design of the revised wing for the present tailless design, few of the usual restrictions were imposed on the aerodynamicist in the determination of wing airfoil section and wing twist.

Inasmuch as the original wing appeared to be subject to premature trailing-edge separation, the airfoil camber-line loading was modified to give as gentle a pressure recovery as possible and still have a specified positive pitching moment at zero lift The leading edge was, therefore, loaded more than was done on the original airfoil, and the load was then distributed more gradually along the chord.

Defendants' Exhibit A—(Continued)

The NACA 63,4-XXX family of airfoils was considered to be best for the present design.

Previous two-dimensional wind-tunnel tests of the original 63,4-221 (a=.27, b=.54) airfoil, proposed as the root section for a tailless design, indicated that there was a correlation factor of about 3 between theoretically calculated section-pitching moment coefficients and those obtained in the wind-tunnel. Examination of theoretical and experimental pressure distribution data indicated that the difference between theory and experiment was greatest near the rear portion of the airfoil.

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This difference between theoretical and experimental pressure distribution data has been examined by Robert M. Pinkerton (Reference 6). Pinkerton explains the difference as an effect of viscosity, which is neglected in the development of the theory. The viscosity of the air is observed as a frictional force producing drag on the airfoil. Since the layer of air that passes over the airfoil is slowed down by this frictional force, a low-energy boundary layer is produced. The boundary layer thickens towards the trailing edge of the airfoil. Since all pressures are transmitted normal to this boundary layer without

Defendants' Exhibit A—(Continued)

change (Pascal's Law), the actual pressure distribution measured over the airfoil is that existing over the contour formed by the boundary layer and not by the material airfoil. The differences between theory and experiment are, therefore, greatest over the aft portion of the airfoil where the boundary layer is thickest and the deviation from the true airfoil contour greatest.

The theoretical pitching-moments of the revised airfoils were, therefore, selected to give one-third the value needed to produce the required wing pitching-moment. The full-scale wing-alone pitching-moment coefficient at zero lift C_{m_0} for proper trim and stability was estimated to be 0.060. Theoretical airfoil pitching-moment coefficients at zero lift of 0.0065 at the root, 0.0195 at the 48% semi-span point, and 0.0325 at the tip were selected as proper values to give this required full-scale wing-alone moment.

Span-load distributions showed that twist distribution alone, as a means of obtaining satisfactory wing stalling characteristics, was isfactory from the viewpoint of drag. [A combination of camber ckness was selected to provide a desirable section maximum-lift- ient distribution and yet to maintain a high critical Mach number. g airfoil sections as selected to meet these criteria are:

3,4-422 (a = .1, b = .59)

i-span: 63,4-318 (a = .1, b = .59)

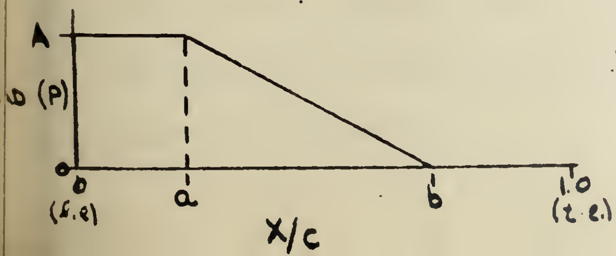
63,4-516 (a = .1, b = .59)

The aerodynamic washout required to obtain favorable stall eristics as well as reasonable drag values was found to be 1° root and splice, with no additional washout between the splice wing tip.]

AIRFOIL THEORY

The characteristic properties of a low-drag airfoil, i.e. the g-moment at zero lift, C_{mo}; the maximum lift-coefficient, $C_{\ell\,max}$; C_ℓ range and location of the minimum-drag region are determined nsiderable extent by the shape of the mean-camber-line, subject fication by the particular thickness distribution of the complete .

The mean-camber-line load distribution can be described by d parameters schematically represented below. The load is as constant from the leading edge to a station "a" on the chord airfoil and is assumed as linearly decreasing to zero from station station "b", the load remaining zero from station "b" to the g-edge of the airfoil.



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The formulae for the ordinates and slope of the camber line from Reference 7, are:

I. Ordinates:

$$y_c = \frac{C_{l_i}}{2\pi(a+b)}\left[\frac{1}{b-a}\left\{\frac{1}{2}(a-x)^2 \ln|a-x| - \frac{1}{2}(b-x)^2 \ln|b-x| + \frac{1}{4}(b-x)^2 - \frac{1}{4}(a-x)^2\right\} - x\ln x + g - hx\right] \quad (1)$$

where:

$$g = -\frac{1}{b-a}\left[a^2\left\{\frac{1}{2}\ln a - \frac{1}{4}\right\} - b^2\left\{\frac{1}{2}\ln b - \frac{1}{4}\right\}\right]$$

$$h = \frac{1}{b-a}\left[\frac{1}{2}(1-a)^2 \ln(1-a) - \frac{1}{2}(1-b)^2 \ln(1-b) + \frac{1}{4}(1-b)^2 - \frac{1}{4}(1-a)^2\right] + g$$

II Slope

$$\frac{dy_c}{dx} = -\frac{C_{l_i}}{2\pi(a+b)}\left[\frac{1}{b-a}\left\{(a-x)\ln|a-x| - (b-x)\ln|b-x|\right\} + \left\{1 + \ln x + h\right\}\right] \quad (2)$$

The design lift coefficient (C_{l_i})* which corresponds closely to t-coefficient for lowest drag, (i.e. the lift coefficient located e center of the low-drag range) is defined as

$$C_{l_i} = \int_0^1 \text{LOAD}\left[d\left(\frac{x}{c}\right)\right] = \int_0^1 P\left[d\left(\frac{x}{c}\right)\right]$$

is the term called ${}_0P_b$ in Reference 1.

$$P = {}_0P_b = \frac{p - p_0}{\frac{1}{2}\rho V_0^2}$$

$$C_{l_i} = \int_0^a A\left[d\left(\frac{x}{c}\right)\right] + \int_a^b \frac{A\left(b-\frac{x}{c}\right)}{(b-a)}\left[d\left(\frac{x}{c}\right)\right] \quad (3)$$

A is the numerical value of the load P in the constant load range).

The term C_{l_b} in Reference 1 is called C_{l_i} using E.N. Jacobs' n in the equation of the camber-line, since Jacobs' notation has common in aeronautical use. C_{l_i} is here defined as the design efficient of the airfoil due to camber, whereas C_{l_b} is the actual lift coefficient of the airfoil and includes the effect of the thickness distribution.

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$$C_{l_i} = Aa + \frac{A}{(b-a)} b(b-a) - \frac{A}{2(b-a)}(b^2 - a^2)$$

$$= A(a+b) - \frac{A(b+a)}{2}$$

$$= \frac{A}{2}(a+b)$$

$$A = \frac{2C_{l_i}}{a+b} \qquad (4)$$

nt coefficient about the aerodynamic center of the airfoil is

$$C_{m_{\delta/c}} = \int_0^1 P(\delta/c - x/c)\, d\left(\frac{x}{c}\right) \qquad (5)$$

$$= \frac{2C_{l_i}}{a+b}\left\{\delta/c\left[\int_0^a d(x/c) + \int_a^b \frac{b-\frac{x}{c}}{b-a} d\left(\frac{x}{c}\right)\right] - \int_0^a \frac{x}{c} d\left(\frac{x}{c}\right) - \int_a^b \frac{b-\frac{x}{c}}{b-a}\left(\frac{x}{c}\right) d\left(\frac{x}{c}\right)\right\}$$

$$= \frac{2C_{l_i}}{a+b}\left\{\frac{\delta}{c}\left[a + b\frac{(b-a)}{(b-a)} - \frac{1}{2}\frac{(b^2-a^2)}{(b-a)}\right] - \left[\frac{1}{2}a^2 + \frac{1}{2}\frac{b(b^2-a^2)}{b-a} - \frac{1}{3}\frac{(b^3-a^3)}{b-a}\right]\right\}$$

$$= \frac{2C_{l_i}}{a+b}\left\{\frac{\delta}{c}\left(\frac{a+b}{2}\right) - \frac{1}{6}\left(a^2 + ab + b^2\right)\right\} \qquad (5a)$$

is the aerodynamic center of the airfoil.

Figure 2 shows the relationship between C_{l_i}, C_{m_0}, "a", and "b" aerodynamic center at 0.265 . This chart can be used as a ary selection chart for "a" and "b" values to give a desired cal C_{m_0} for a given design lift coefficient C_{l_i}.

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GENERAL THEORY OF AIRFOIL PRESSURE DISTRIBUTIONS

The pressure distributions for the airfoils treated in this were computed by the method outlined in References 1 and 2.

The principle involved is that of obtaining the pressure e velocity assuming a non-viscous, incompressible fluid in ;ional motion,

lli's equation is

$$H = p + \frac{1}{2}\rho V^2$$

Neglecting compressibility and viscosity the pressure at any of the upper surface is:

$$p_u = H - \frac{1}{2}\rho V_u^2$$

$= p_0 + \frac{1}{2}\rho V_0^2$ for some point in the free stream far from the l.

$$p_u - p_0 = \frac{1}{2}\rho\left[V_0^2 - V_u^2\right]$$

$$\frac{p_u - p_0}{\frac{1}{2}\rho V_0^2} = 1 - \left(\frac{V_u}{V_0}\right)^2$$

rly for the lower surface

$$\frac{p_l - p_0}{\frac{1}{2}\rho V_0^2} = 1 - \left(\frac{V_l}{V_0}\right)^2$$

ad on the airfoil is then

$$P = \frac{p_l - p_u}{\frac{1}{2}\rho V_0^2} = \left(\frac{V_u}{V_0}\right)^2 - \left(\frac{V_l}{V_0}\right)^2$$

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C_{l_b} is the basic lift due to the shape of the camber line and
ess distribution and C_{l_a} is the additional lift due to the change
le of attack*, the incremental load changes for airfoil of small
ess can be written.

$$\frac{V_u}{V_o} = \frac{V_s}{V_o} + \frac{\Delta u}{V_o} C_{l_b} + \frac{\Delta V_a}{V_o} C_{l_a}$$

$$\frac{V_l}{V_o} = \frac{V_s}{V_o} - \frac{\Delta u}{V_o} C_{l_b} - \frac{\Delta V_a}{V_o} C_{l_a}$$

$\frac{V_s}{V_o}$ is the velocity at any point on the symmetrical airfoil of the
hickness,

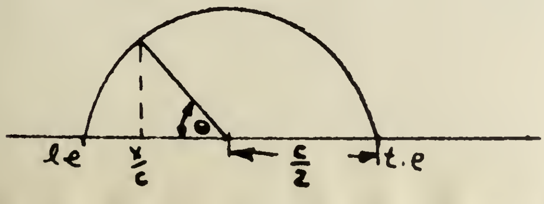
$\frac{\Delta u}{V_o} C_{l_b}$ is the velocity increment due to the camber line
g at C_{l_b}, and $\frac{\Delta V_a}{V_o} C_{l_a} = \frac{\Delta V_a}{V_o}(C_l - C_{l_b})$ is the velocity increment
symmetrical airfoil due to the change in angle of attack required
ain C_{l_a}. Substituting these values of velocity into the equations
e load

$$P = \left(\frac{V_u}{V_o}\right)^2 - \left(\frac{V_l}{V_o}\right)^2$$

It has been found (Reference 3) that the lift of airfoils of
thickness can be divided into two parts. One part is due only
shape of the airfoil camber line, and the other is due to the
of attack of the airfoil measured from an angle of attack, α_i,

$$\alpha_i = \frac{1}{\pi}\int_0^{\pi} \frac{dy_c}{d\left(\frac{x}{c}\right)} d\theta$$

$\frac{x}{c} = \frac{1}{2}(1 - \cos\theta)$ as graphically represented below:



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$$\left(\frac{V_s}{V_o}+\frac{\Delta u}{V_o}C_{l_b}+\frac{\Delta V_a}{V_o}C_{l_a}\right)^2-\left(\frac{V_s}{V_o}-\frac{\Delta u}{V_o}C_{l_b}-\frac{\Delta V_a}{V_o}C_{l_a}\right)^2$$

$$=\frac{4V_s}{V_o}\left[\frac{\Delta u}{V_o}C_{l_b}+\frac{\Delta V_a}{V_o}C_{l_a}\right]$$

re,

$$=\int_0^1\frac{4V_s}{V_o}\left[\frac{\Delta u}{V_o}C_{l_b}+\frac{\Delta V_a}{V_o}C_{l_a}\right]\left[d\left(\frac{x}{c}\right)\right]$$

$$\int_0^1\frac{4V_s}{V_o}\left[\frac{\Delta u}{V_o}C_{l_b}+\frac{\Delta V_a}{V_o}C_{l_a}\right]\left[\frac{\delta}{c}-\frac{x}{c}\right]\left[d\left(\frac{x}{c}\right)\right]$$

$_2 = 0$

$$=\int_0^1\frac{4V_s}{V_o}\left[\frac{\Delta u}{V_o}C_{l_b}+\frac{\Delta V_a}{V_o}C_{l_a}\right]\left[d\left(\frac{x}{c}\right)\right]$$

$$m_o=-\int_0^1\frac{4V_s}{V_o}\left[\frac{\Delta u}{V_o}C_{l_b}+\frac{\Delta V_a}{V_o}C_{l_a}\right]\left[\frac{x}{c}\right]\left[d\left(\frac{x}{c}\right)\right]$$

he following method is employed to obtain C_{m_o}:

C_{l_i} is obtained by integrating the slope of the camber-line

$$\left(\frac{4u}{V_o}C_{l_i}\right)_{\theta=\theta_o}=-\frac{1}{2\pi}\int_0^{2\pi}\frac{dy_c}{d(x/c)}\cot(\theta-\theta_o)\,d\theta$$

his integral can be evaluated by a numerical method given in

ces 1 and 4 and outlined in Appendix A of this report.

$_b$ is obtained by integrating the load due to C_{l_i}.

$$C_{l_b}=\int_0^1\frac{4V_s}{V_a}\left(\frac{\Delta u}{V_o}C_{l_i}\right)\left[d\left(\frac{x}{c}\right)\right]$$

aking the area between the pressure distribution curves $\frac{p_l-p_o}{1/2\rho V_o^2}$

$\frac{p_u-p_o}{1/2\rho V_o^2}$

$_a$ is set equal to $-C_{l_b}$ so obtained in order to get $C_l=0$

general C_{l_b} is not equal to C_{l_i}.)

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C_{m_0} is computed from the integral

$$C_{m_0} = -\int_0^1 \frac{4V_s}{V_0}\left[\frac{\Delta u}{V_0} C_{l_b} + \frac{\Delta V_a}{V_0} C_{l_a}\right]\left[\frac{x}{c}\right]\left[d\left(\frac{x}{c}\right)\right]$$

TABLE OF NOMENCLATURE AND DEFINITIONS

= Basic sectional lift coefficient. The basic lift depends on the airfoil camber line and the thickness distribution, and is independent of angle of attack.

= Additional sectional lift coefficient. The additional lift depends only on the angle of attack as measured from α_i

= Angle of attack at which the additional lift is zero.

= Sectional design lift coefficient at which the additional lift of airfoil is zero. This lift coefficient occurs very close to the center of the minimum drag region. $C_{\ell_i} = C_{\ell_b}$ for airfoils of infinitely small thickness.

= Camber line load parameters, expressed in percent of chord.

= Aerodynamic center about which pitching moment is taken

= Sectional pitching moment.

= Basic load; identical to ${}_oP_b$ in Reference 1.

p = static pressure at a point of the airfoil contour

p_0 = static pressure in free stream

V_0^2 = dynamic pressure in free stream.

θ = the angle whose cosine is $2\frac{x}{c} - 1$ (expressed in radians)

x/c = any point on the chord of the airfoil

= ratio of incremental velocity on airfoil to free stream velocity.

ript

u denotes upper surface

ℓ denotes lower surface

s, t denotes thickness

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$\frac{\Delta u}{V_0} C_{l_i}$ OR $\frac{\Delta u}{V_0} C_{l_b}$ is the velocity due to the camber camber line and thickness respectively,

$\frac{\Delta V_a}{V_0} C_{l_a}$ = velocity on the symmetrical airfoil due to the change in angle of attack required to obtain C_{l_a}.

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Distribution; by Theodore Theodoresen; NACA TR 383.

eral Potential Theory of Arbitrary Wing Sections; by T. Theodorsen
I. E. Garrick; NACA T.R. 452.

-1 -(1) "Spanwise Air Load Distribution"

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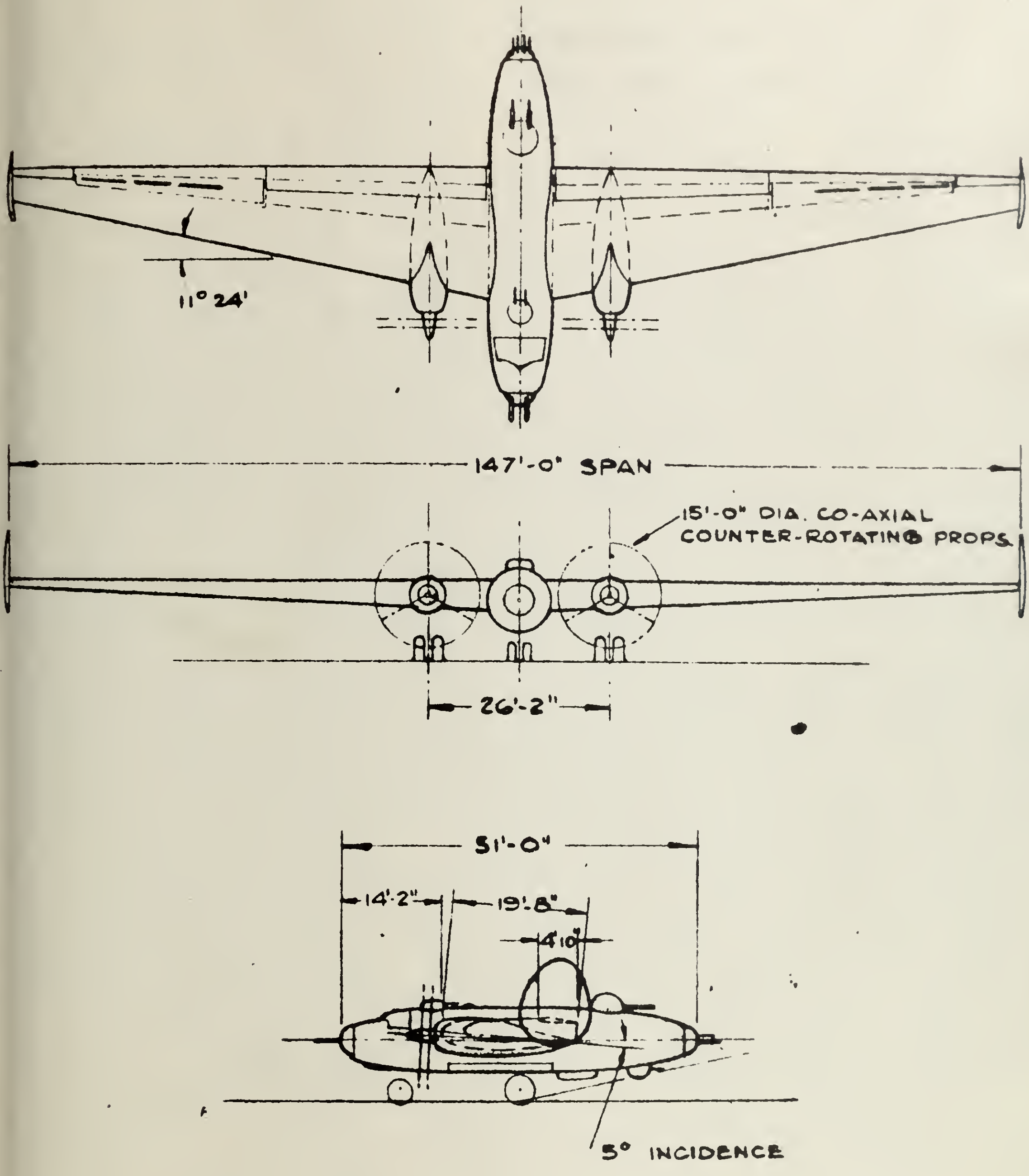
FIG 1



SCALE - 1/300

NUÑEZ	2-8-44	GENERAL ARRANGEMENT REVISED 2-ENGINE TAILLESS DESIGN	PART NUMBER
MR	2-8-44	CONSOLIDATED AIRCRAFT CORPORATION LINDBERGH FIELD • SAN DIEGO, CALIFORNIA	

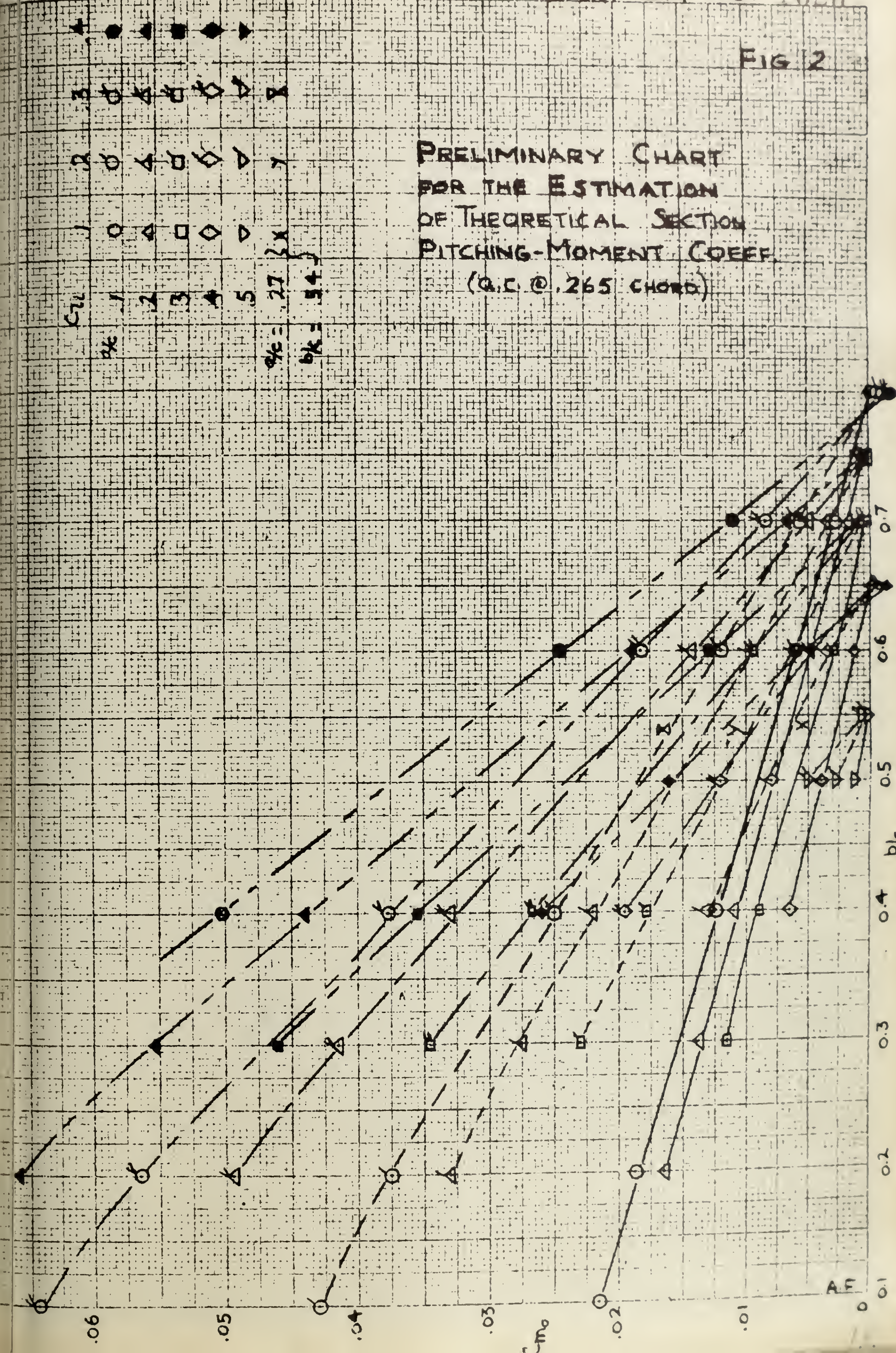
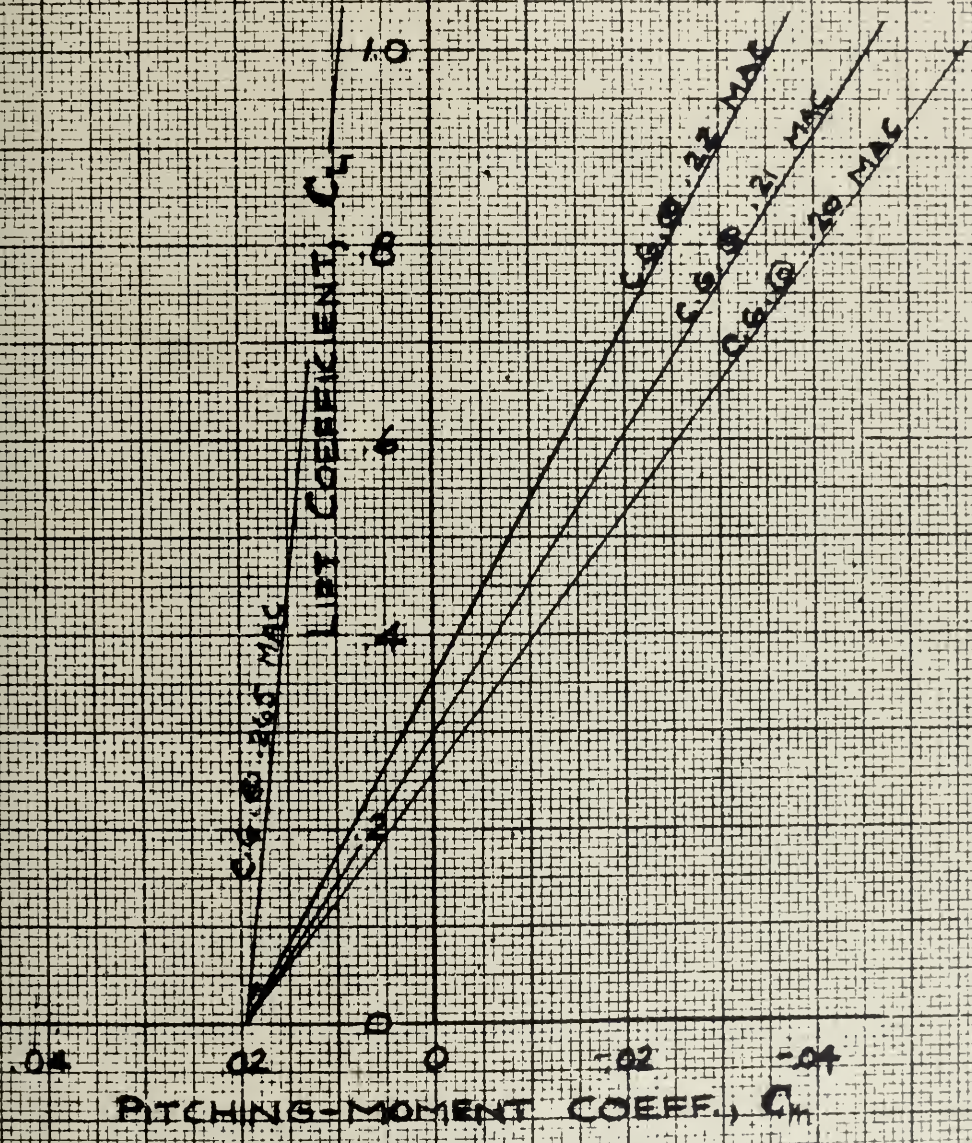


FIG 2

PRELIMINARY CHART FOR THE ESTIMATION OF THEORETICAL SECTION PITCHING-MOMENT COEFF. (A.C. @ .265 CHORD)

FIG. 3

REVISED WING
ESTIMATED PITCHING-MOMENT COEFF.
COMPLETE MODEL
(NO POWER)



NOTE: ΔC_m = -.04 ESTIMATED FOR FUSELAGE, NACELLES, ETC.

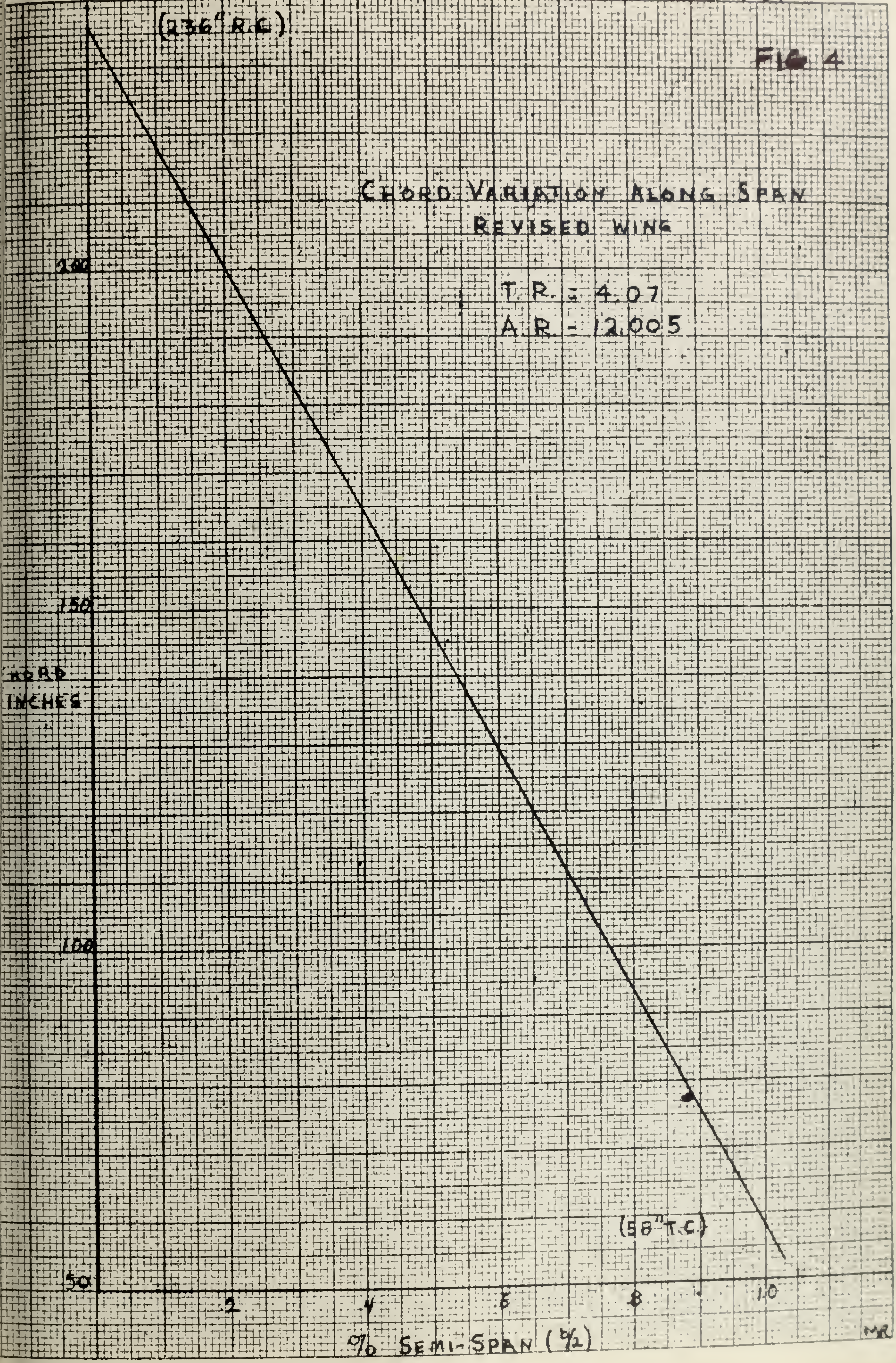
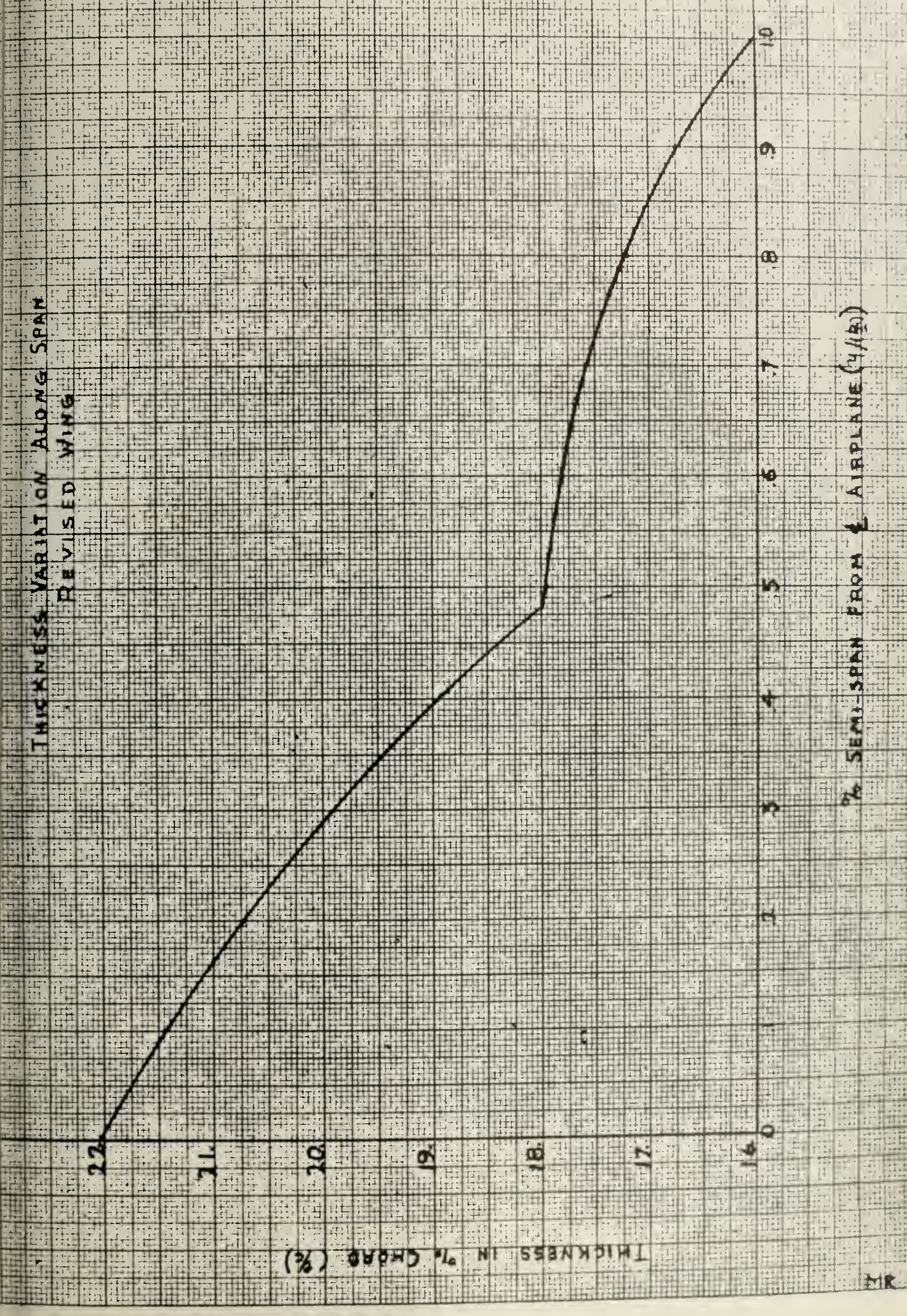

(236" R.C)
CHORD VARIATION ALONG SPAN
REVISED WING
T.R. = 4.07
A.R. = 12.005
200
150
CHORD
INCHES
100
50
(58" T.C.)
.2
.4
.6
.8
1.0
% SEMI-SPAN (b/2)
MR

FIG 4

FIG 5


THICKNESS VARIATION ALONG SPAN
REVISED WING
% SEMI-SPAN FROM ℄ AIRPLANE (y/(b/2))
0
1
2
3
4
5
6
7
8
9
10
THICKNESS IN % CHORD (%)
22
21
20
19
18
17
16
MR

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…nt Engineering - San Diego, California

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FIG. 6

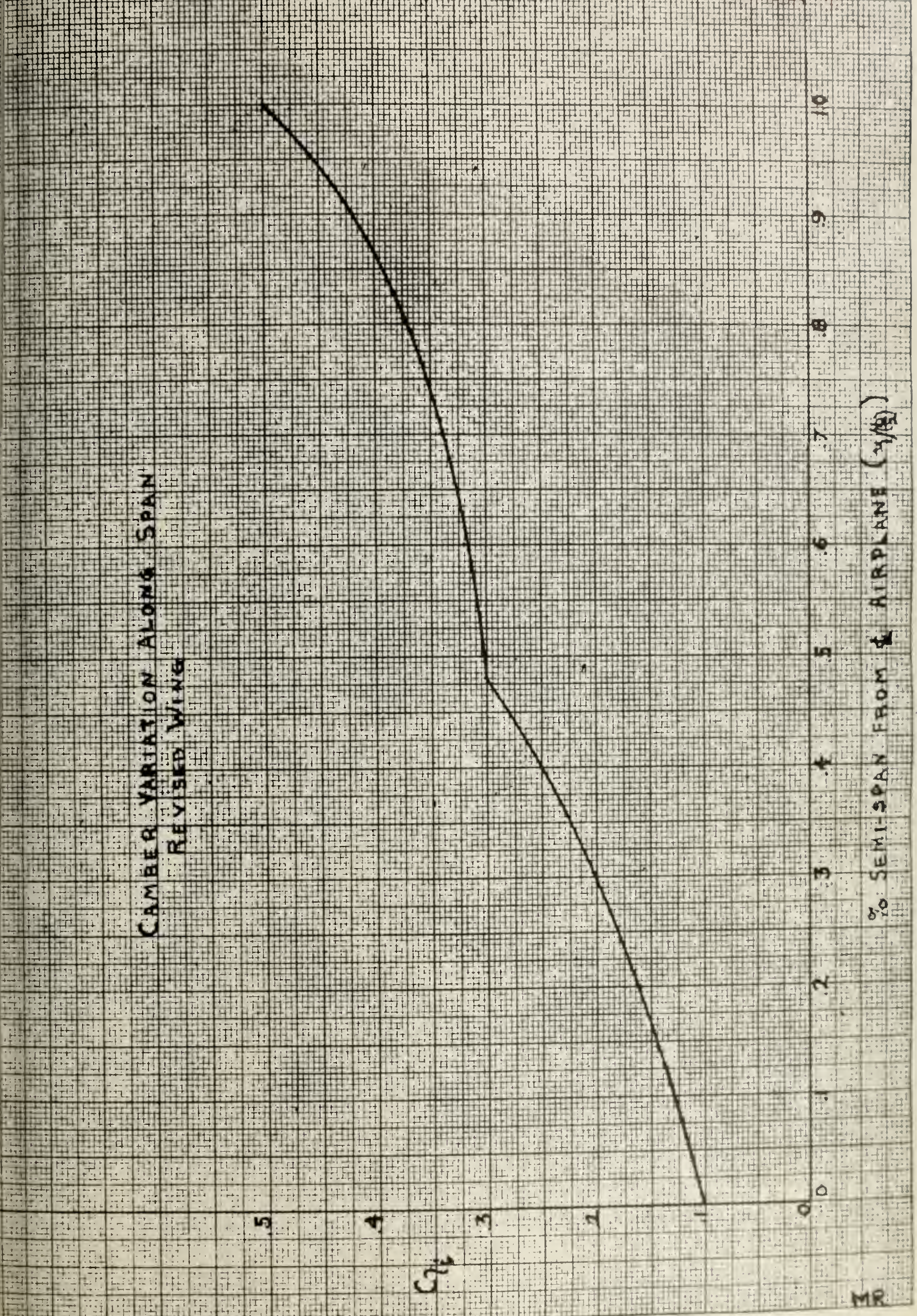
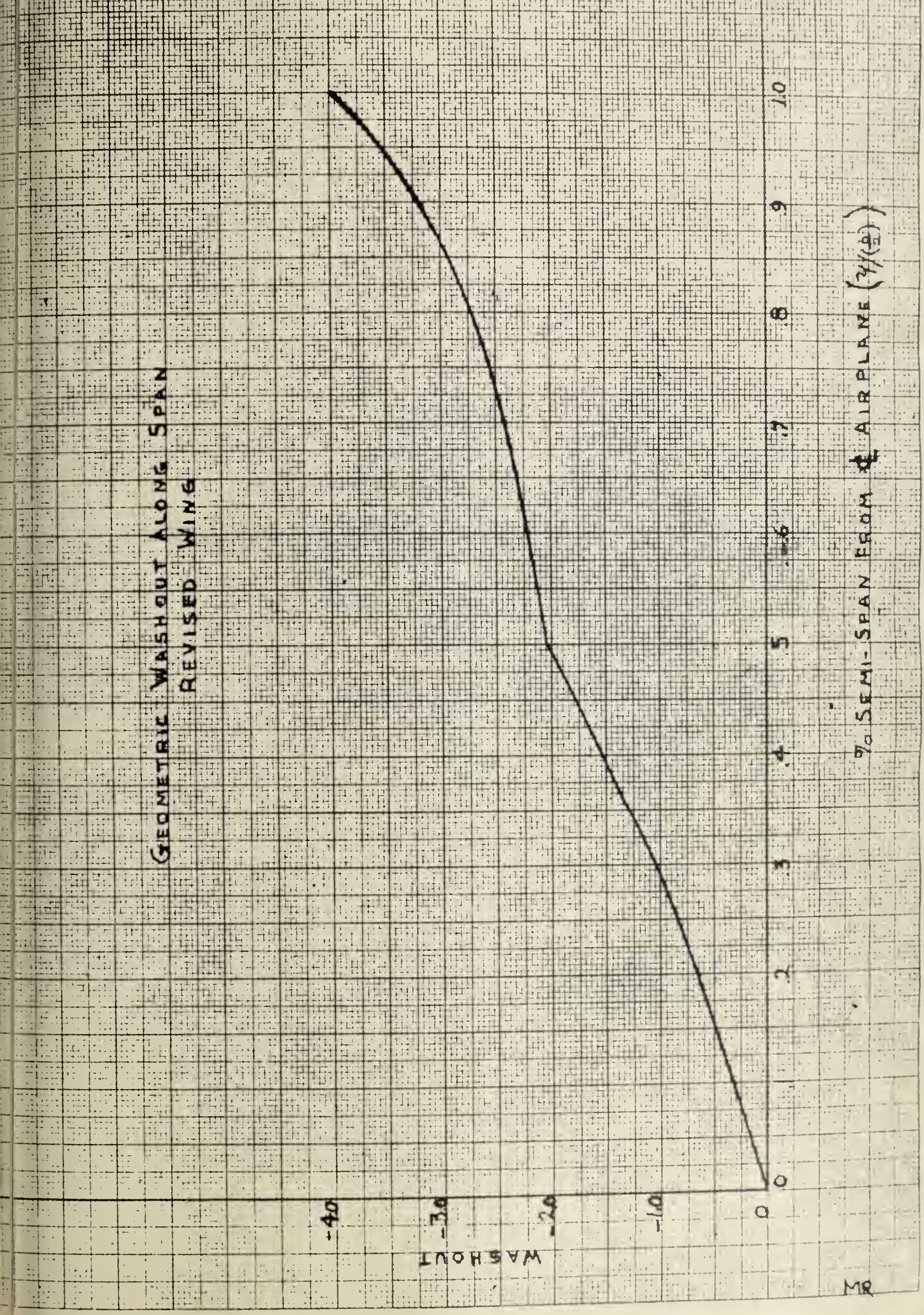


FIG 7


GEOMETRIC WASHOUT ALONG SPAN
REVISED WING
% SEMI-SPAN FROM ℄ AIRPLANE (y/(b/2))
0
.1
.2
.3
.4
.5
.6
.7
.8
.9
1.0
WASHOUT
0
-1.0
-2.0
-3.0
-4.0
MR

APPENDIX A

OUTLINE OF PROCEDURE FOR THE CALCULATION OF AIRFOIL PRESSURE DISTRIBUTIONS

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OUTLINE OF PROCEDURE FOR THE CALCULATION OF AIRFOIL PRESSURE DISTRIBUTIONS

s can be seen from the general theory there are three velocities rmine: $\frac{V_s}{V_0}$, the velocity due to the airfoil shape; $\frac{\Delta u}{V_0}$, the y increment due to camber (or to basic lift); and $\frac{\Delta V_a}{V_0}$, the y increment due to angle of attack (or to additional lift).

$$\frac{V_s}{V_0} = \left(\frac{V}{V_0} - 1\right)\frac{t}{t_b} + 1$$

re $\frac{V}{V_0}$ is the velocity on the base profile as given in Reference 2. uation merely gives the correction to the velocity for thicknesses than those of Reference 2.

To quote a numerical example (not related to the subject airfoils): t desired to determine the velocity on the surface of the symmetrical l 63,4-022 at 0.25c

$$\left(\frac{V_s}{V_0}\right)_{0.25c} = (1.288 - 1)\frac{22}{20} + 1 = 1.317$$

This is also the velocity increment due to airfoil thickness at the e of a cambered airfoil at the same station.

$\frac{\Delta u}{V_0} C_{l_i}$ is the velocity increment due to the camber line. Since the line can be replaced by a vortex sheet, the velocities will add on p of the airfoil and subtract on the bottom for positive lift.

$\frac{\Delta u}{V_0} C_{l_i} = \frac{P}{4}$ where P is the load at any point on the camber line. be approximated by the theoretical load, but should be obtained by ating the camber line slope:

$$\left(\frac{\Delta u}{V_0} C_{l_i}\right)_{\theta=\theta_0} = \left(\frac{P}{4}\right)_{\theta=\theta_0} = -\frac{1}{2\pi}\int_0^{2\pi}\left(\frac{dy_c}{d(x/c)}\right)\cot(\theta-\theta_0)\,d\theta$$

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s ntegral can be evaluated numerically by the method given in 1 and 4 where $F = dy_c/d(x/c)$ in Reference 1).

procedure to be used is as follows:

Plt $dy_c/d(x/c)$ vs θ (not vs. $\frac{x}{c}$)

Fa r the plot of $dy_c/d(x/c)$ vs. θ so that it intersects the line $\theta = 0$ (Theoretically $dy_c/d(x/c) = \infty$ at $\theta = 0$). This is illustrated in Fig. 3.

a ational method of estimating $\left(dy_c/d(x/c)\right)_{0,0}$ is shown in Appendix D.

Ke d off values of $dy_c/d(x/c)$ and $\frac{d}{d\theta}\left(dy_c/d(x/c)\right)$ at $\theta = 0$, 0.1π, 0.2π, ...
0.9π, π .

Se these values into the formula

$$\left(\frac{\Delta u}{V_0} C_{l_i}\right)_{\theta=\theta_0} = -\left\{a_0\left[\frac{d}{d\theta}\left(\frac{dy_c}{d(x/c)}\right)\right]_{\theta=\theta_0} + a_1\left[\left(dy_c/d(x/c)\right)_{\theta=\theta_0+0.1\pi} - \left(dy_c/d(x/c)\right)_{\theta=\theta_0-0.1\pi}\right]\right.$$
$$\left. + \dots\dots + a_9\left[\left(dy_c/d(x/c)\right)_{\theta=\theta_0+0.9\pi} - \left(dy_c/d(x/c)\right)_{\theta=\theta_0-0.9\pi}\right]\right\}$$

numerical values of the coefficients a_n are:

a_0 = 0.1000	a_5 = 0.0503
a_1 = 0.3473	a_6 = 0.0366
a_2 = 0.1572	a_7 = 0.0281
a_3 = 0.0996	a_8 = 0.0163
a_4 = 0.0691	a_9 = 0.0080

$\Delta V_a/V_0 \, C_{l_a}$ is the velocity increment due to the circulation round a symmetrical airfoil lifting at $C_l = C_{l_a}$

($C_{l_a} = C_l - C_{l_b}$ where $C_{l_b} = 0$ for a symmetrical airfoil)

BY

CHECKED

APPROVED

umerical values for $\left(\frac{\Delta V_a}{V_o}\right)_{C_{\ell_a}=1}$ are given in the same tables
hose for $\frac{V}{V_o}$. For any other value of C_{ℓ_a} multiply the given
ue f $\left(\frac{\Delta V_a}{V_o}\right)_{C_{\ell_a}=1}$ by the desired value of C_{ℓ_a} .

BY
CHECKED
APPROVED

FIG 1

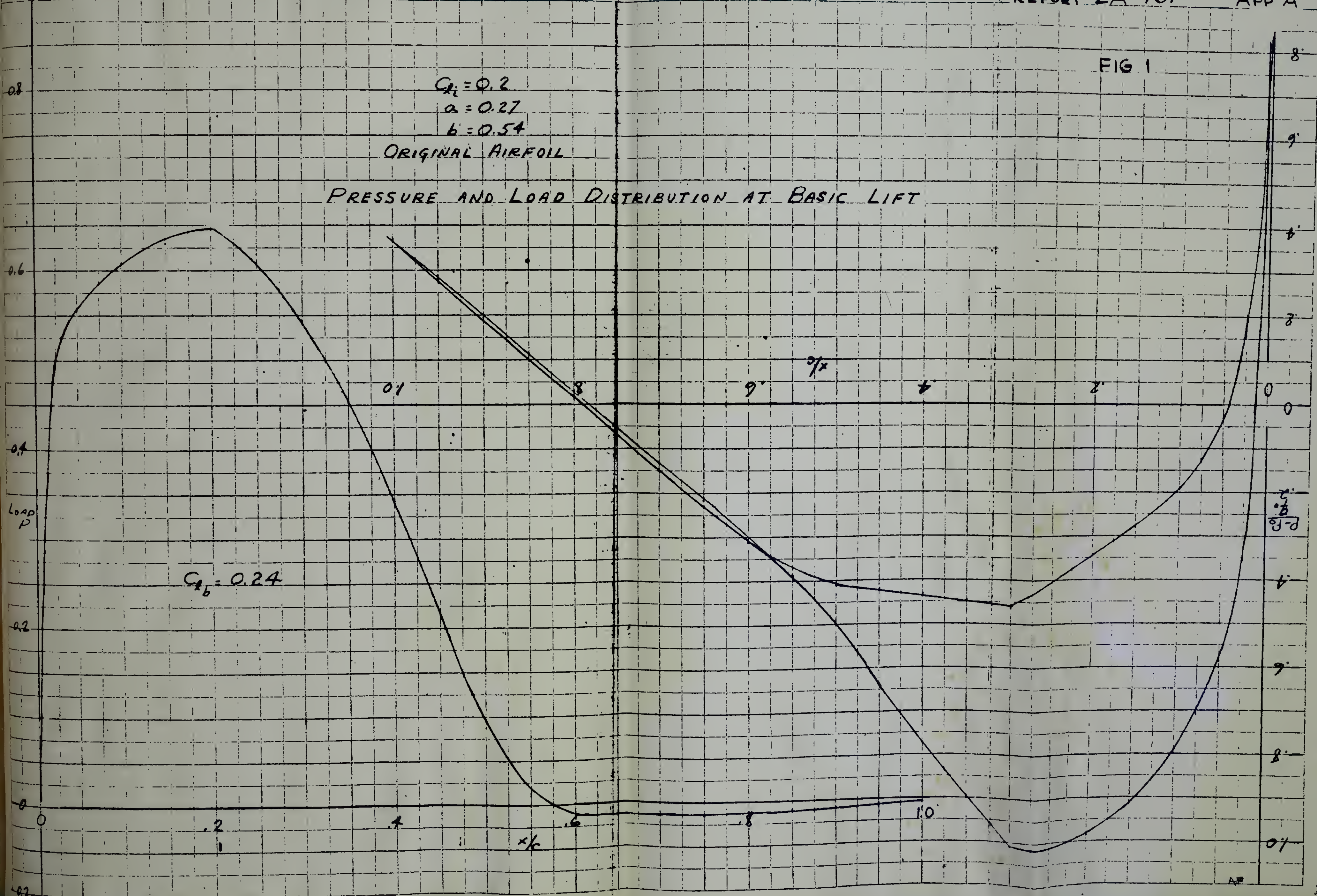

$C_{l_i} = 0.2$
$a = 0.27$
$b = 0.54$
ORIGINAL AIRFOIL
PRESSURE AND LOAD DISTRIBUTION AT BASIC LIFT
$C_{l_b} = 0.24$
LOAD P
x/c
y/x
0.8
0.6
0.4
0.2
0
.2
.4
.6
.8
1.0

FIG 2

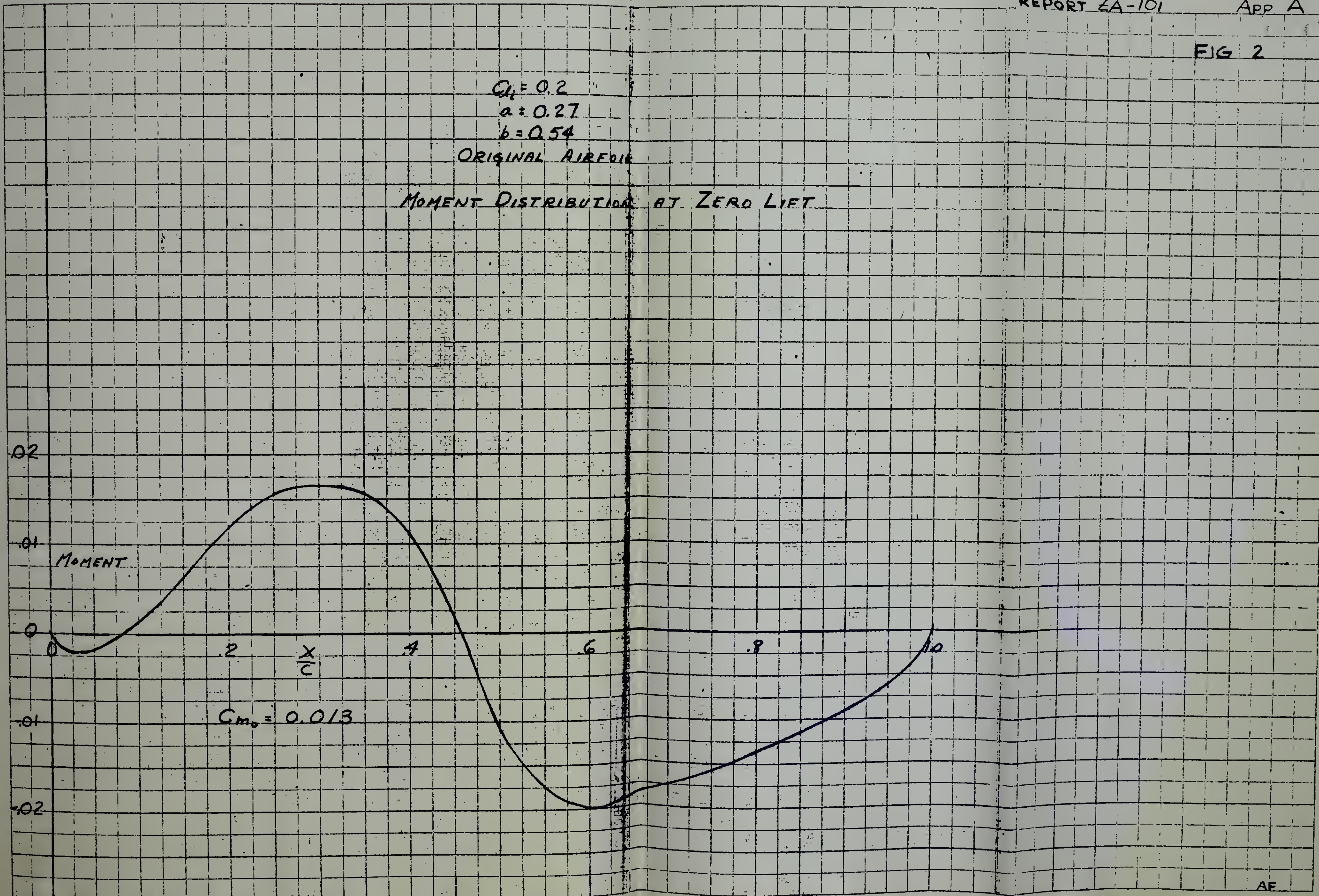
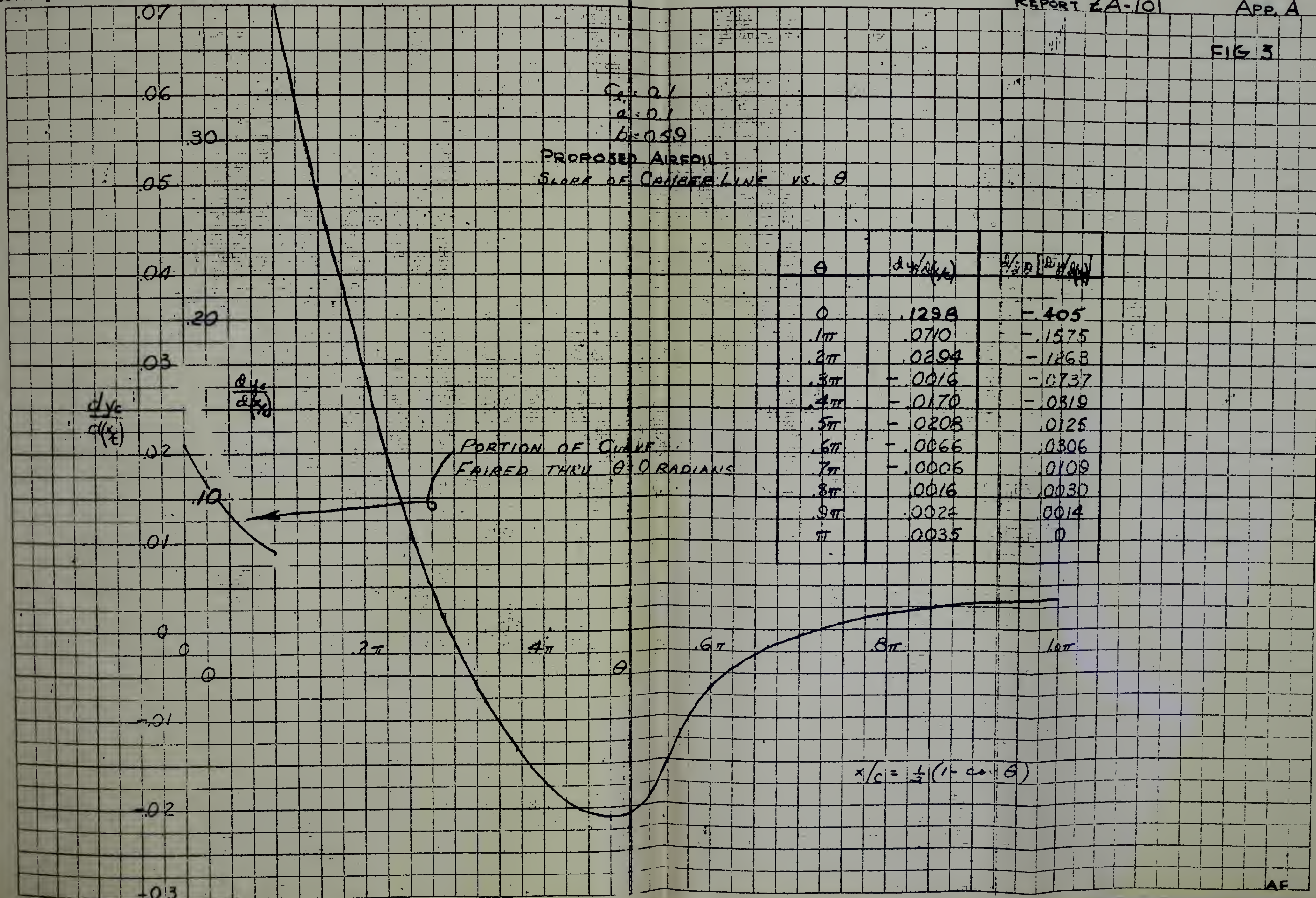

$C_{l_i} = 0.2$
$a = 0.27$
$b = 0.54$
ORIGINAL AIRFOIL
MOMENT DISTRIBUTION AT ZERO LIFT
.02
.01
MOMENT
0
-.01
-.02
0
.2
x/c
.4
.6
.8
1.0
$C_{m_0} = 0.013$
AF

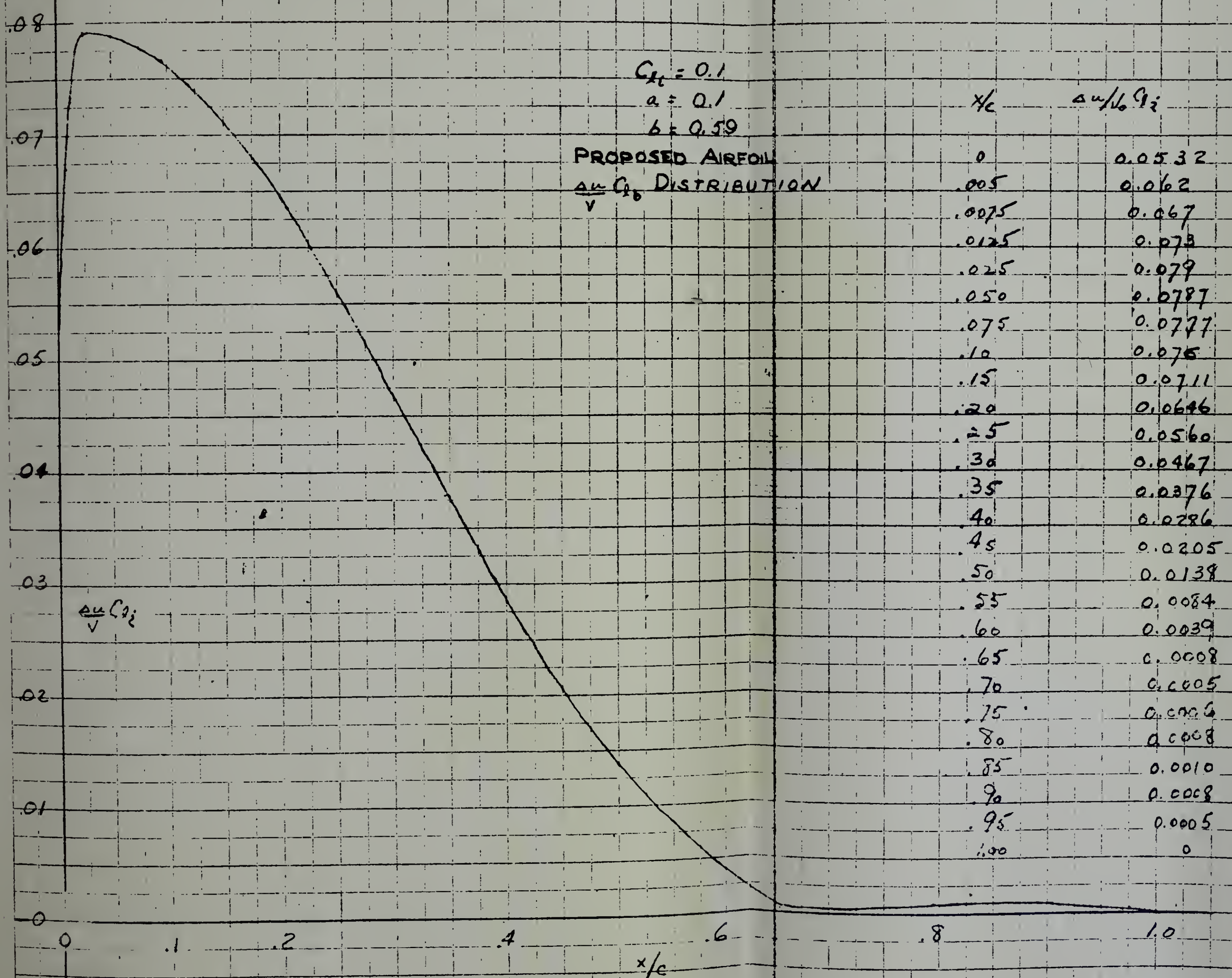
FIG 3



θ	$dy_c/d(x/c)$	$\frac{d}{d\theta}\left[dy_c/d(x/c)\right]$
0	.1298	-.405
.1π	.0710	-.1575
.2π	.0294	-.1268
.3π	-.0016	-.0737
.4π	-.0170	-.0319
.5π	-.0208	.0125
.6π	-.0066	.0306
.7π	-.0006	.0109
.8π	.0016	.0030
.9π	.0024	.0014
π	.0035	0

AF

FIG 4



x/c	$\Delta u / V_0 C_{l_i}$
0	0.0532
.005	0.062
.0075	0.067
.0125	0.073
.025	0.079
.050	0.0787
.075	0.0777
.10	0.076
.15	0.0711
.20	0.0646
.25	0.0560
.30	0.0467
.35	0.0376
.40	0.0286
.45	0.0205
.50	0.0138
.55	0.0084
.60	0.0039
.65	0.0008
.70	0.0005
.75	0.0006
.80	0.0008
.85	0.0010
.90	0.0008
.95	0.0005
1.00	0

AF

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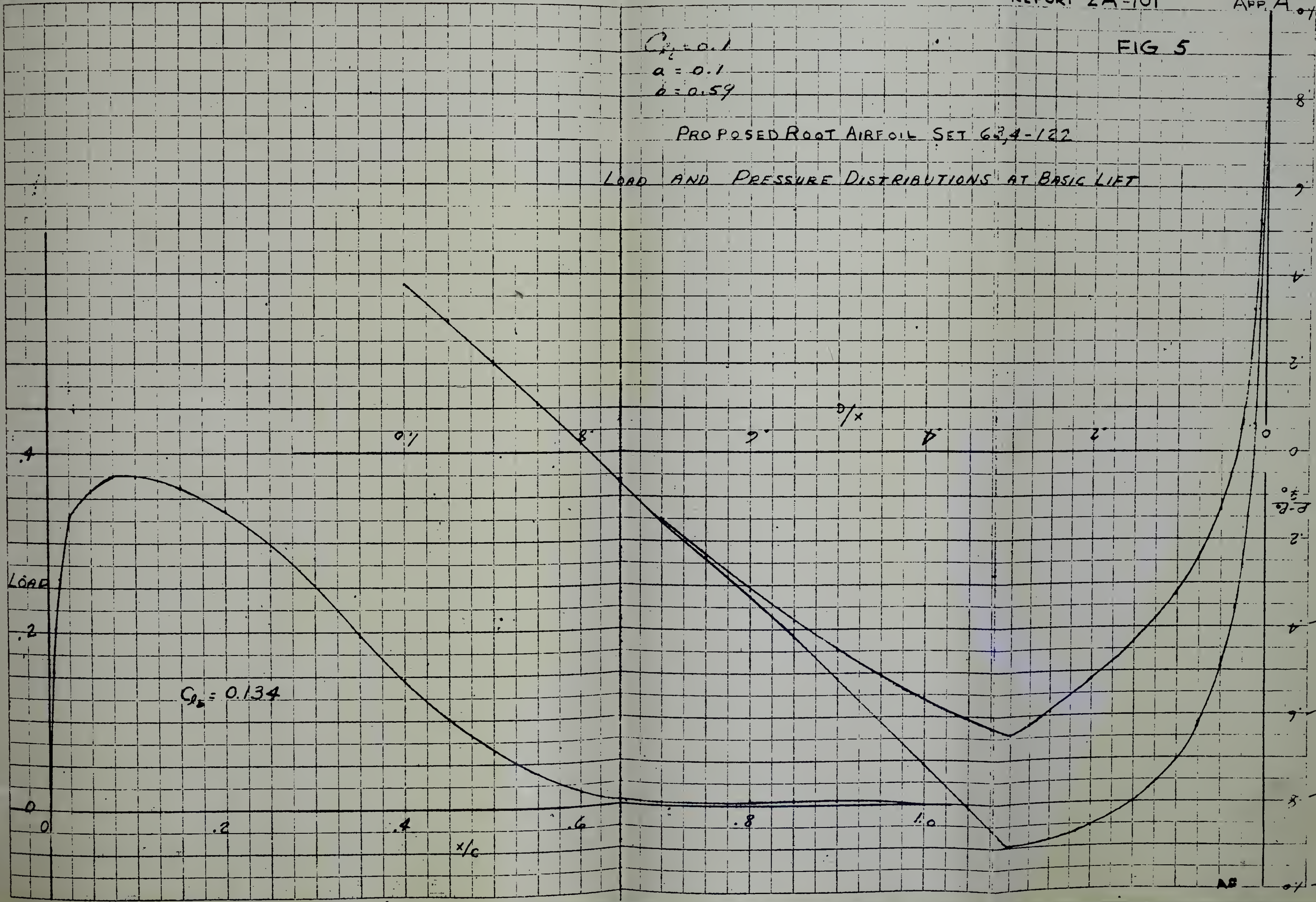

FIG 5

FIG 6

$C_{l_i} = 0.1$
$a = 0.1$
$b = 0.59$

PROPOSED ROOT AIRFOIL SET, 63,4-122

MOMENT DISTRIBUTION

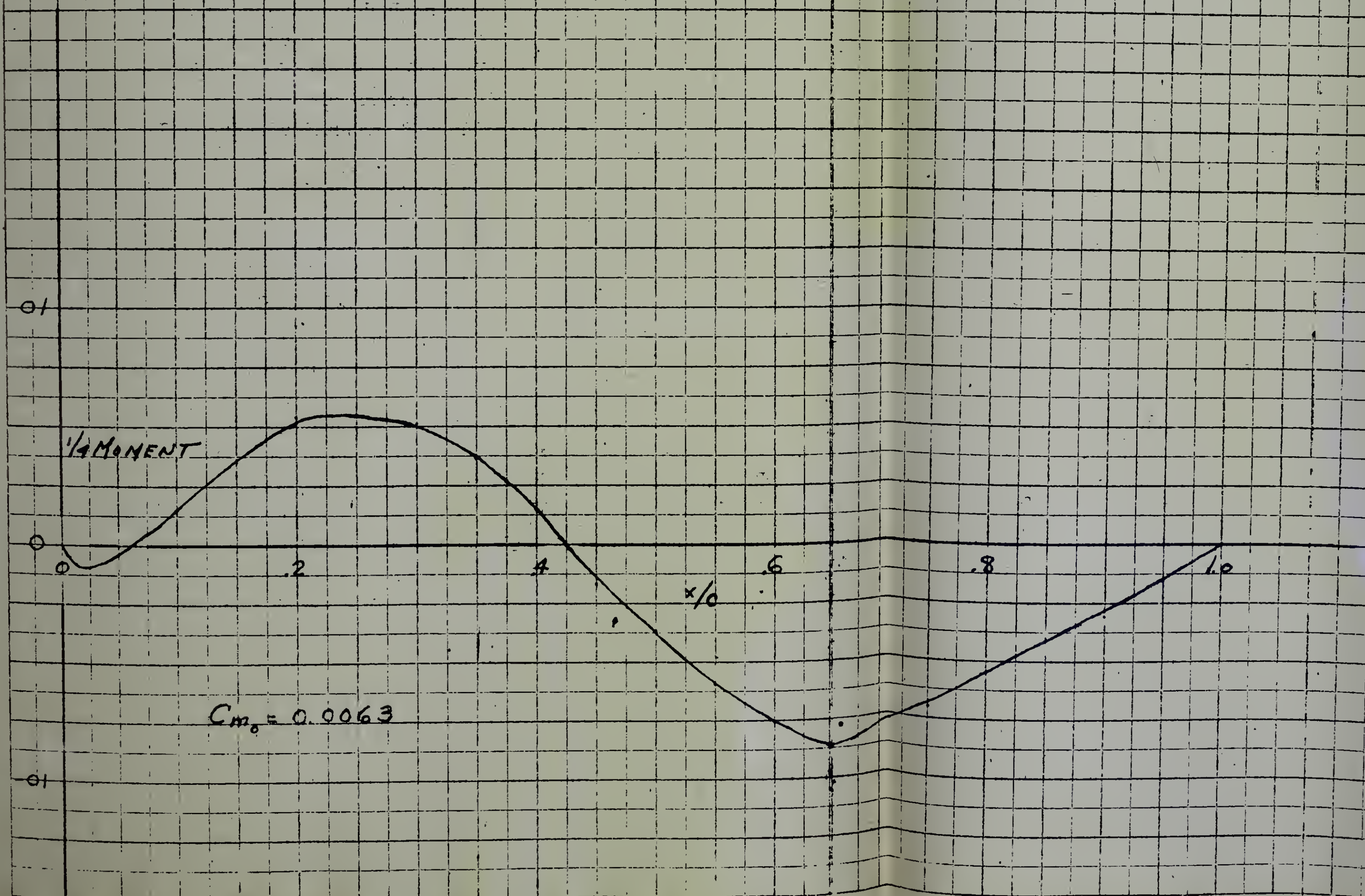

FIG 7

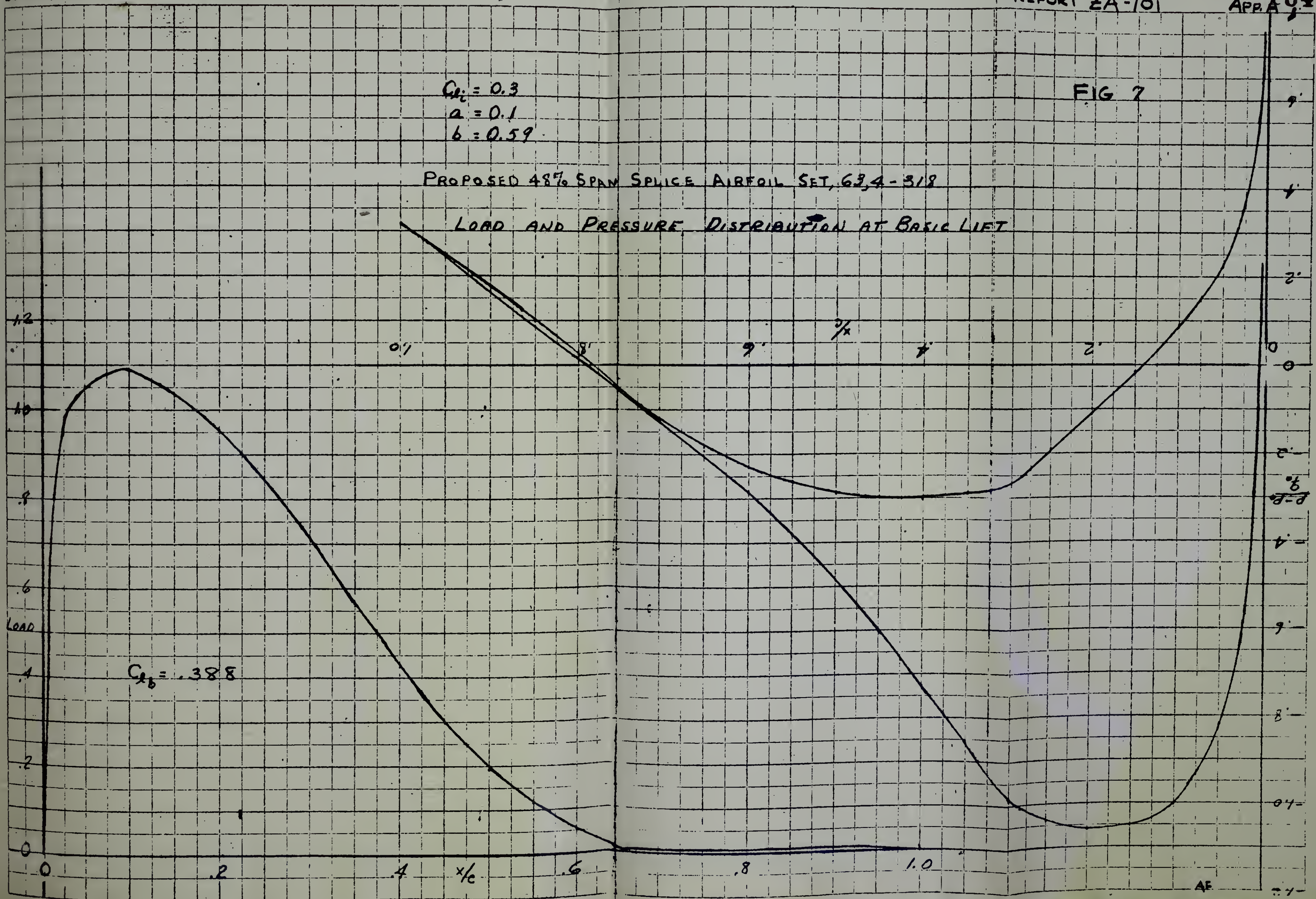
$C_{l_i} = 0.3$
$a = 0.1$
$b = 0.59$
PROPOSED 48% SPAN SPLICE AIRFOIL SET, 63,4-318
LOAD AND PRESSURE DISTRIBUTION AT BASIC LIFT
$C_{l_b} = .388$
LOAD
1.2
1.0
.8
.6
.4
.2
0
0
.2
.4
x/c
.6
.8
1.0

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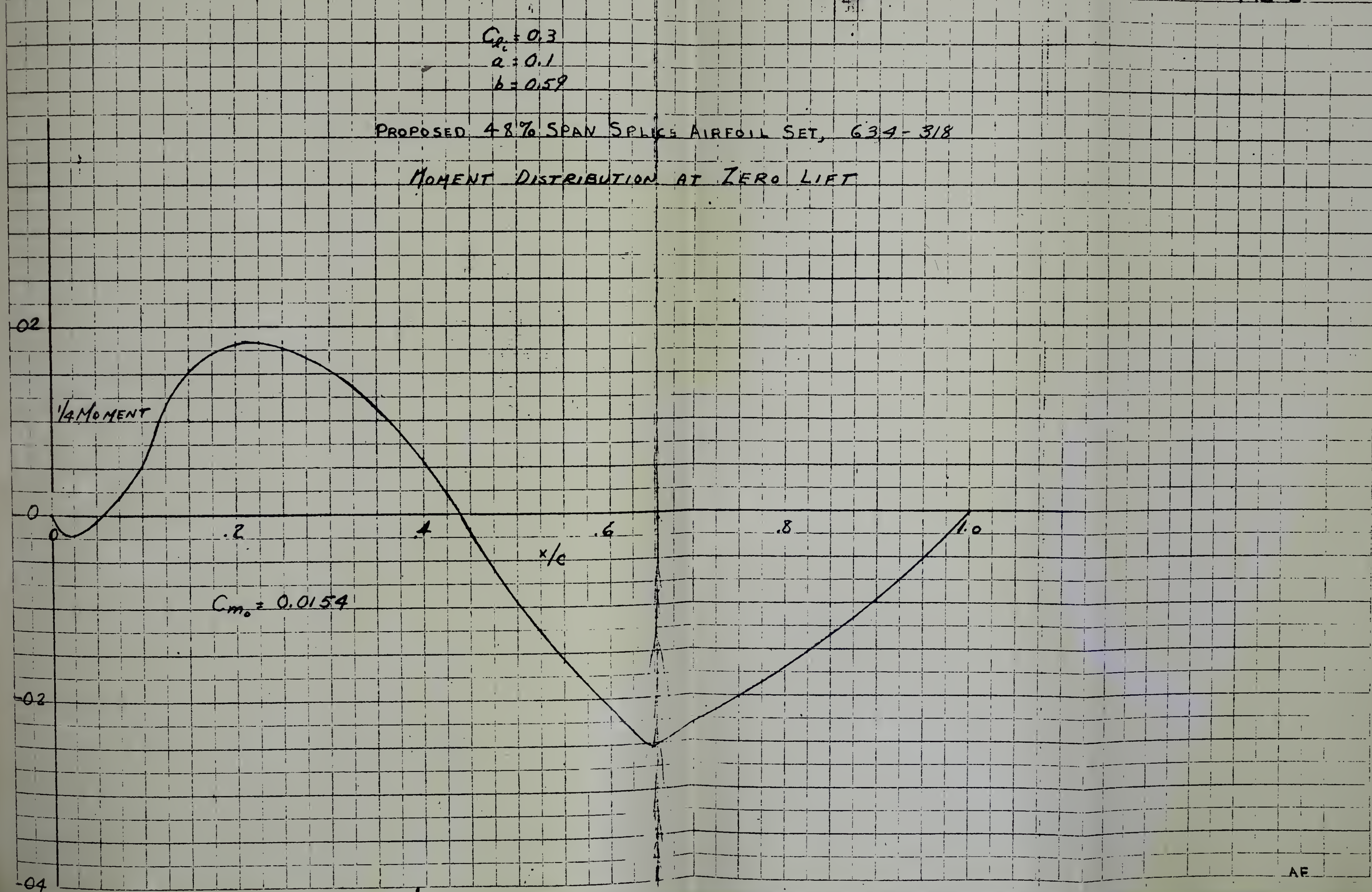
FIG 8



AF

FIG. 9

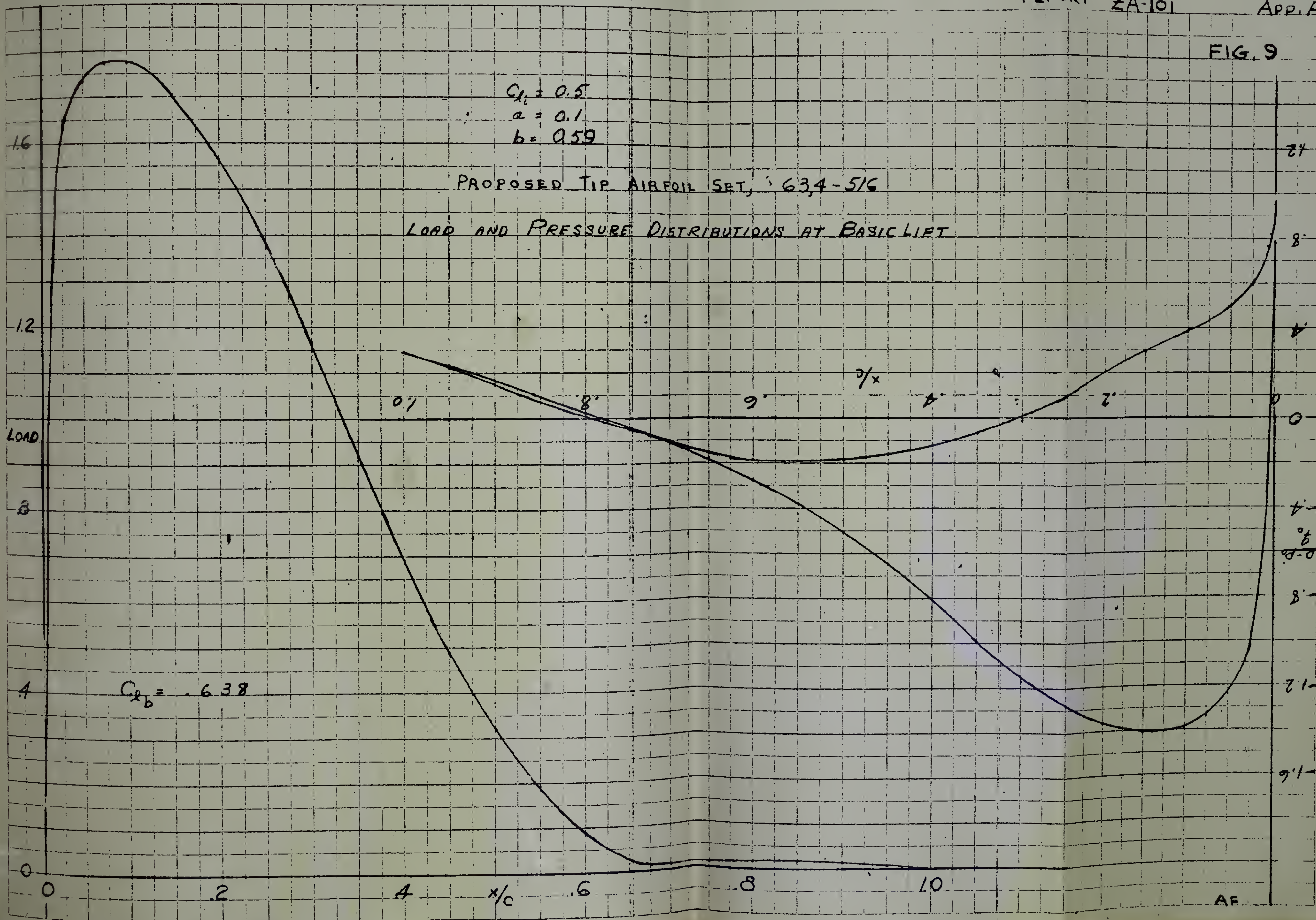
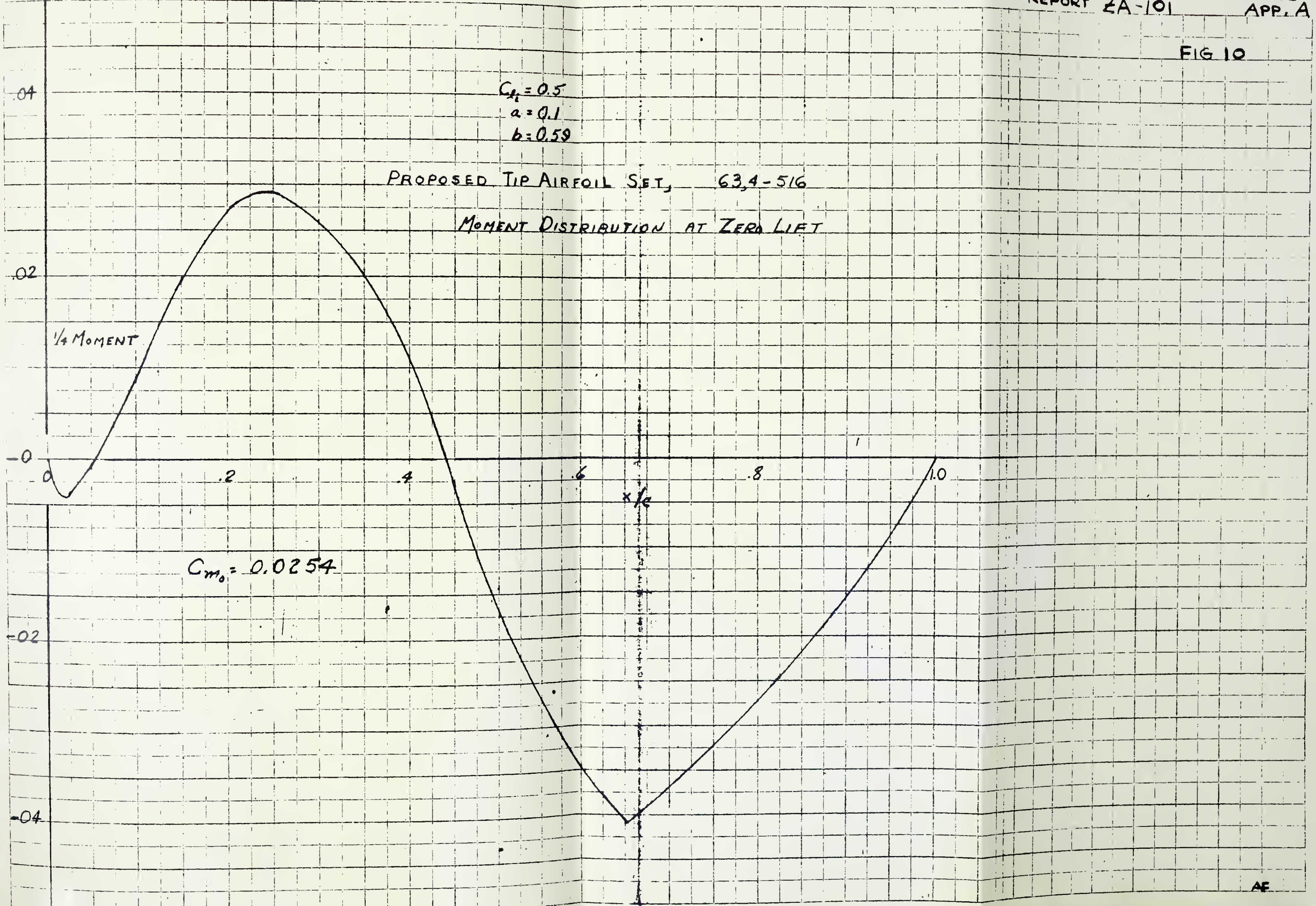

$C_{l_i} = 0.5$
$a = 0.1$
$b = 0.59$
PROPOSED TIP AIRFOIL SET, 63,4-516
LOAD AND PRESSURE DISTRIBUTIONS AT BASIC LIFT
LOAD
$C_{l_b} = .638$
1.6
1.2
.4
0
.2
.4
x/c
.6
.8
1.0
AF

FIG 10


$C_{l_i} = 0.5$
$a = 0.1$
$b = 0.59$
PROPOSED TIP AIRFOIL SET, 63,4-516
MOMENT DISTRIBUTION AT ZERO LIFT
1/4 MOMENT
.04
.02
-0
-.02
-.04
0
.2
.4
.6
.8
1.0
x/c
$C_{m_0} = 0.0254$

AF

INCREMENTAL VELOCITY DUE TO CAMBER

$C_{l_i} = 0.1 \quad (a = 0.1,\ b = 0.59)$

θ	dy_c/dx	$\frac{d(dy_c/dx)}{d\theta}$	1 a_n	$\theta=0$: 2	$\theta=0$: 3	$\theta=0$: 4 (3)-(2)	$\theta=0$: 5 (1)(4)	$\theta=.1\pi$: 2	$\theta=.1\pi$: 3	$\theta=.1\pi$: 4 (3)-(2)	$\theta=.1\pi$: 5 (1)(4)	$\theta=.2\pi$: 2	$\theta=.2\pi$: 3	$\theta=.2\pi$: 4 (3)-(2)	$\theta=.2\pi$: 5 (1)(4)
0	.1665	-.532	0.1	-.532		.532	.0532	-.1575		.1575	.0158	-.1263		.1263	.0126
.1π	.0710	-.1575	.3473			0	0	.0294	.1665	.1371	.0476	-.0016	.0710	.0726	.0252
.2π	.0294	-.1263	.1592					-.0016	.0710	.0726	.0114	-.0170	.1665	.1835	.0288
.3π	-.0016	-.0737	.0996					-.0170	.0294	.0464	.0046	-.0208	.0710	.0918	.0092
.4π	-.0170	-.0319	.0691					-.0208	-.0016	.0192	.0013	-.0066	.0294	.0360	.0025
.5π	-.0208	.0125	.0503					-.0066	-.0170	-.0104	-.0005	-.0006	-.0016	-.0010	-.00005
.6π	-.0066	.0306	.0366					-.0006	-.0208	-.0202	-.0007	.0016	-.0170	-.0186	-.00068
.7π	-.0006	.0109	.0281					.0016	-.0066	-.0082	-.0002	.0026	-.0208	-.0234	-.00066
.8π	.0016	.0020	.0163					.0026	-.0006	-.0032	-.00005	.0034	-.0066	-.0100	-.00016
.9π	.0026	.0014	.0080					.0034	.0016	-.0018	-.00001	.0026	-.0006	-.0032	-.00002
π	.0034	0													

$\theta = 0$: $\frac{\Delta u}{V_0} = .0532$

$\theta = .1\pi$: $\frac{\Delta u}{V_0} = +.0793$

$\theta = .2\pi$: $\frac{\Delta u}{V_0} = +.0767$

1	$\theta=.3\pi$: 2	$\theta=.3\pi$: 3	$\theta=.3\pi$: 4	$\theta=.3\pi$: 5	$\theta=.4\pi$: 2	$\theta=.4\pi$: 3	$\theta=.4\pi$: 4	$\theta=.4\pi$: 5	$\theta=.5\pi$: 2	$\theta=.5\pi$: 3	$\theta=.5\pi$: 4	$\theta=.5\pi$: 5
.1	-.0737		.0737	.0074	-.0319		.0319	.0032	.0125		-.0125	-.0013
.3473	-.0170	.0294	.0464	.0162	-.0208	-.0016	.0192	.0067	-.0066	-.0170	-.0104	-.0036
.1572	-.0208	.0710	.0918	.0144	-.0066	.0294	.0360	.0057	-.0006	-.0016	-.0010	-.00016
.0996	-.0066	.1665	.1731	.0173	-.0006	.0710	.0716	.0071	.0016	.0294	+.0278	+.0028
.0691	-.0006	.0710	.0716	.0050	.0016	.1665	.1649	.0114	.0026	.0710	.0684	.0047
.0503	.0016	.0294	.0278	.0014	.0026	.0710	.0684	.0034	.0034	.1665	.1631	.0082
.0366	.0026	-.0016	-.0042	-.0002	.0034	.0294	.0260	.0010	.0026	.0710	.0684	.0025
.0281	.0034	-.0170	-.0204	-.0006	.0026	-.0016	-.0042	-.0001	.0016	.0294	.0278	.00078
.0163	.0026	-.0208	-.0234	-.0004	.0016	-.0170	-.0186	-.0003	-.0006	-.0016	-.0010	.00001
.0080	.0016	-.0066	-.0082	-.0001	-.0006	-.0208	-.0202	-.0002	-.0066	-.0170	-.0164	.00008

$\theta = .3\pi$: $\frac{\Delta u}{V_0} = +.0634$

$\theta = .4\pi$: $\frac{\Delta u}{V_0} = +.0379$

$\theta = .5\pi$: $\frac{\Delta u}{V_0} = +.0138$

TABLE II

INCREMENTAL VELOCITY DUE TO CAMBER, $\frac{\Delta u}{V_0}$

$C_{l_i} = 0.1 \quad (a=0.1, b=0.59)$

	$\theta = .6\pi$				$\theta = .7\pi$				$\theta = .8\pi$			
1	2	3	4	5	2	3	4	5	2	3	4	5
.1	.0306		-.0306	-.0031	.0109		-.0109	-.0011	.0030		-.0030	-.0003
.3473	-.0006	-.0208	-.0202	-.0070	.0016	-.0066	-.0082	-.0028	.0026	-.0006	-.0032	-.0011
.1572	.0016	-.0170	-.0186	-.0029	.0026	-.0208	-.0234	-.0037	.0034	-.0066	-.0100	-.0016
.0996	.0026	-.0016	-.0042	-.0004	.0034	-.0170	-.0204	-.0020	.0026	-.0208	-.0234	-.0023
.0691	.0034	.0294	+.0260	+.0018	.0026	-.0016	-.1042	-.0003	.0016	-.0170	-.0186	-.0013
.0503	.0026	.0710	.0634	.0034	.0016	.0294	+.0278	+.0014	-.0006	-.0016	-.0010	-.00035
.0366	.0016	.1665	.1649	.0060	-.0006	.0710	.0716	.0026	-.0066	.0294	+.0360	+.0013
.0281	-.0006	.0710	.0716	.0020	-.0066	.1665	.1731	.0049	-.0208	.0710	.0918	.0026
.0163	-.0066	.0294	.0360	.00059	-.0208	.0710	.0918	.0015	-.0170	.1665	.1835	.0030
.0080	-.0208	-.0016	.0192	.00015	-.0170	.0294	.0464	.00037	-.0016	.0720	.0726	.00058

$\theta = .6\pi$: $\frac{\Delta u}{V_0} = +.0005$

$\theta = .7\pi$: $\frac{\Delta u}{V_0} = +.0009$

$\theta = .8\pi$: $\frac{\Delta u}{V_0} = +.0008$

	$\theta = .9\pi$			
1	2	3	4	5
.1	.0014		-.0014	-.0001
.3473	.0034	.0016	-.0018	-.0006
.1572	.0026	-.0006	-.0032	-.0005
.0996	.0016	-.0066	-.0082	-.0008
.0691	-.0006	-.0208	-.0202	-.0014
.0503	-.0066	-.0170	-.0104	-.0005
.0366	-.0208	-.0016	+.0192	+.0007
.0281	-.0170	.0294	.0464	.0019
.0163	-.0016	.0710	.0726	.0012
.0080	.0294	.1665	.1371	.0011

$\frac{\Delta u}{V_0} = +.0004$

$\theta = \pi$

1			
0		0	0

$\frac{\Delta u}{V_0} = 0$

$$\left(\frac{\Delta u}{V_0} C_{l_i}\right)_{\theta=\theta_0} = -\left\{(a_0)\left[\frac{d}{d\theta}\left(\frac{dy_c}{d(\frac{x}{c})}\right)\right]_{\theta=\theta_0} + (a_1)\left[\left(\frac{dy_c}{d(\frac{x}{c})}\right)_{\theta=\theta_0+0.1\pi} - \left(\frac{dy_c}{d(\frac{x}{c})}\right)_{\theta=\theta_0-.1\pi}\right] + (a_2)\left[\left(\frac{dy_c}{d(\frac{x}{c})}\right)_{\theta=\theta_0+0.2\pi} - \left(\frac{dy_c}{d(\frac{x}{c})}\right)_{\theta=\theta_0-0.2\pi}\right] + \ldots\ldots + (a_9)\left[\left(\frac{dy_c}{d(\frac{x}{c})}\right)_{\theta=\theta_0+0.9\pi} - \left(\frac{dy_c}{d(\frac{x}{c})}\right)_{\theta=\theta_0-0.9\pi}\right]\right\}$$

CALCULATION OF PRESSURE DISTRIBUTION
PROPOSED ROOT SECTION
63,4-122 (a=0.1, b=0.59)

$C_{\ell_b} = .134$

1 x/c	2 v/V_0	3 (2)-1	4 1.1(3)	5 v_t/V_0 (4)+1	6 $\Delta v/V_0\, C_{\ell_i}$	7 (5)+(6)	8 $(7)^2$	9 P_u 1-(8)	10 (5)-(6)	11 $(10)^2$	12 P_ℓ 1-(11)	13 LOAD (12)-(9)	14 $(\Delta v_a/V_0)$ × (.134)	15 (6)-(14)	16 1/4 LOAD (5)(15)	17 1/4 MOM (1)(16)
0	0	-1.000	-1.100	-.100	.053	-.047	.002	.998	-.153	.023	.977	-.021	.187	-.134	.0134	0
.0050	.666	-.334	-.368	.632	.062	.694	.482	.518	.570	.325	.675	.157	.172	-.110	-.0695	-.0004
.0075	.778	-.222	-.244	.756	.067	.823	.678	.322	.689	.475	.525	.203	.161	-.094	-.0711	-.0005
.0125	.906	-.094	-.104	.896	.073	.969	.938	.062	.823	.678	.322	.260	.144	-.071	-.0636	-.0008
.0250	1.039	.039	.043	1.043	.079	1.122	1.260	-.260	.964	.929	.071	.331	.113	-.034	-.0355	-.0009
.0500	1.130	.130	.143	1.143	.079	1.222	1.492	-.492	1.064	1.133	-.133	.359	.086	-.007	-.0080	-.0004
.0750	1.176	.176	.194	1.194	.078	1.272	1.620	-.620	1.116	1.244	-.244	.376	.073	.005	.0060	.0005
.1000	1.207	.207	.228	1.228	.076	1.304	1.702	-.702	1.152	1.328	-.328	.374	.064	.012	.0147	.0015
.1500	1.245	.245	.270	1.270	.071	1.341	1.800	-.800	1.199	1.436	-.436	.364	.052	.019	.0241	.0036
.2000	1.270	.270	.297	1.297	.065	1.362	1.856	-.856	1.232	1.519	-.519	.337	.044	.021	.0272	.0054
.2500	1.288	.288	.317	1.317	.056	1.373	1.882	-.882	1.261	1.590	-.590	.292	.039	.017	.0224	.0055
.3000	1.300	.300	.330	1.330	.047	1.377	1.899	-.899	1.283	1.645	-.645	.254	.024	.013	.0173	.0052
.3500	1.277	.277	.305	1.305	.037	1.342	1.802	-.802	1.268	1.607	-.607	.195	.029	.008	.0105	.0037
.4000	1.252	.252	.277	1.277	.029	1.306	1.704	-.704	1.248	1.556	-.556	.148	.026	.003	.0038	.0015
.4500	1.225	.225	.247	1.247	.020	1.267	1.607	-.607	1.227	1.505	-.505	.102	.023	-.003	-.0037	-.0017
.5000	1.197	.197	.217	1.217	.014	1.231	1.513	-.513	1.203	1.450	-.450	.063	.020	-.006	-.0073	-.0037
.5500	1.167	.167	.184	1.184	.008	1.192	1.420	-.420	1.176	1.381	-.381	.039	.017	-.009	-.0107	-.0059
.6000	1.135	.135	.149	1.149	.004	1.153	1.328	-.328	1.145	1.311	-.311	.017	.015	-.011	-.0126	-.0076
.6500	1.101	.101	.111	1.111	.001	1.112	1.236	-.236	1.110	1.232	-.232	.004	.013	-.012	-.0133	-.0087
.7000	1.066	.066	.073	1.073	.001	1.074	1.155	-.155	1.072	1.150	-.150	.005	.011	-.010	-.0107	-.0075
.7500	1.029	.029	.032	1.032	.001	1.033	1.066	-.066	1.031	1.062	-.062	.004	.010	-.009	-.0093	-.0070
.8000	.989	-.011	-.012	.988	.001	.989	.978	.022	.987	.974	.974	.004	.008	-.007	-.0067	-.0055
.8500	.947	-.053	-.058	.942	.001	.943	.892	.108	.941	.885	.885	.007	.006	-.005	-.0047	-.0040
.9000	.901	-.099	-.109	.891	.001	.892	.795	.205	.890	.792	.792	.003	.005	-.004	-.0036	-.0032
.9500	.853	-.147	-.162	.838	.001	.839	.704	.296	.837	.700	.700	.004	.003	-.002	-.0017	-.0016
1.0000	.807	-.193	-.212	.788	0	.788	.622	.378	.788	.622	.622	0	0	0	0	0

$$v_t/V_0 = (v/V_0 - 1)\frac{t}{t_b} + 1$$

$$t/t_b = \frac{0.22}{0.20} = 1.10$$

TABLE IV

CALCULATION OF PRESSURE DISTRIBUTION
PROPOSED AIRFOIL @ 0.48 SEMI-SPAN
63,4-318 (a=0.1, b=0.59)

$C_{l_b} = 0.388$

1	2	3	4	5	6	7	8	9	10	11	12	13	14	15	16	17
x/c	v/V_0	(2)-1	0.9(3)	V_t/V_0 (4)+1	$\Delta v/V_0\, C_{l_i}$	(5)+(6)	$(7)^2$	P_u 1-(8)	(5)-(6)	$(10)^2$	P_l 1-(11)	LOAD (12)-(9)	$(\Delta v_a/V_0) \times (.388)$	(6)-(14)	1/4 LOAD (5)(15)	1/4 MOM. (1)(16)
0	0	-1.000	-.900	.100	.160	.260	.067	.933	-.060	.004	.996	.063	.542	-.382	-.038	0
.0050	.666	-.334	-.301	.699	.186	.885	.783	.217	.513	.264	.736	.519	.497	-.311	-.218	-.0011
.0075	.778	-.222	-.200	.800	.201	1.001	1.002	-.002	.599	.359	.641	.643	.467	-.266	-.213	-.0016
.0125	.906	-.094	-.085	.915	.219	1.134	1.289	-.289	.696	.485	.515	.804	.416	-.197	-.180	-.0020
.0250	1.039	.039	.035	1.035	.237	1.272	1.620	-.620	.798	.638	.362	.982	.328	-.091	-.094	-.0023
.0500	1.130	.130	.117	1.117	.236	1.353	1.830	-.830	.881	.777	.223	1.053	.250	-.014	-.016	-.0008
.0750	1.176	.176	.158	1.158	.233	1.391	1.935	-.935	.925	.855	.145	1.080	.211	.022	.025	.0019
.1000	1.207	.207	.186	1.186	.228	1.414	2.000	-1.000	.958	.919	.091	1.091	.184	.044	.052	.0052
.1500	1.245	.245	.220	1.220	.213	1.433	2.050	-1.050	1.007	1.014	-.014	1.036	.150	.063	.077	.0155
.2000	1.270	.270	.243	1.243	.194	1.437	2.062	-1.062	1.049	1.100	-.100	.962	.128	.066	.082	.0164
.2500	1.288	.288	.259	1.259	.168	1.427	2.040	-1.040	1.091	1.190	-.190	.850	.112	.056	.071	.0177
.3000	1.300	.300	.270	1.270	.140	1.410	1.990	-.990	1.130	1.278	-.278	.712	.100	.040	.051	.0153
.3500	1.277	.277	.249	1.249	.113	1.362	1.855	-.855	1.136	1.288	-.288	.567	.085	.028	.035	.0123
.4000	1.252	.252	.227	1.227	.086	1.313	1.727	-.727	1.141	1.303	-.303	.424	.074	.012	.015	.0060
.4500	1.225	.225	.202	1.202	.062	1.264	1.601	-.601	1.140	1.300	-.300	.301	.066	-.064	-.005	-.0023
.5000	1.197	.197	.177	1.177	.041	1.218	1.482	-.482	1.136	1.290	-.290	.192	.057	-.016	-.019	-.0095
.5500	1.167	.167	.150	1.150	.025	1.175	1.381	-.381	1.125	1.265	-.265	.116	.050	-.025	-.029	-.0160
.6000	1.135	.135	.121	1.121	.012	1.133	1.282	-.282	1.109	1.230	-.230	.052	.043	-.031	-.035	-.0210
.6500	1.101	.101	.091	1.091	.002	1.093	1.192	-.192	1.089	1.185	-.185	.007	.038	-.036	-.039	-.0254
.7000	1.066	.066	.059	1.059	.002	1.061	1.126	-.126	1.057	1.119	-.119	.007	.033	-.031	-.033	-.0231
.7500	1.029	.029	.026	1.026	.002	1.028	1.057	-.057	1.024	1.050	-.050	.007	.028	-.026	-.027	-.0202
.8000	.989	-.011	-.010	.990	.002	.992	.983	.017	.988	.976	.024	.007	.023	-.021	-.021	-.0168
.8500	.947	-.053	-.048	.952	.003	.955	.913	.087	.949	.901	.099	.012	.018	-.015	-.014	-.0119
.9000	.901	-.099	-.089	.911	.002	.913	.835	.165	.909	.826	.174	.009	.014	-.012	-.011	-.0099
.9500	.853	-.147	-.132	.868	.002	.870	.757	.243	.866	.750	.250	.007	.009	-.007	-.006	-.0057
1.0000	.807	-.193	-.174	.826	0	.826	.682	.318	.826	.682	.318	0	0	0	0	0

$$V_t/V_0 = (v/V_0 - 1)\frac{t}{t_b} + 1$$

$$t/t_b = \frac{0.18}{0.20} = 0.90$$

CALCULATION OF PRESSURE DISTRIBUTION
PROPOSED TIP SECTION
63,4-516 $(a=0.1, b=0.59)$

TABLE V

$C_{l_b} = 0.638$

1	2	3	4	5	6	7	8	9	10	11	12	13	14	15	16	17
x/c	V/V_0	$V/V_0 - 1$	0.8(3)	V_t/V 1+(4)	$\frac{\Delta v}{V_0} C_{l_i}$	(5)+(6)	$(7)^2$	P_u 1-(8)	(5)-(6)	$(10)^2$	P_ℓ 1-(11)	LOAD (12)-(9)	$(\Delta V/V_0) \times (.638)$	(6)-(14)	1/4 LOAD (5)(15)	1/4 MOM. (1)(16)
0	0	-1.	-.800	.200	.266	.466	.217	.783	-.066	.004	.996	.213	.891	-.625	-.125	0
.0050	.666	-.334	-.267	.733	.310	1.043	1.090	-.090	.423	.180	.820	.910	.817	-.507	-.372	-.0019
.0075	.778	-.222	-.178	.822	.335	1.157	1.338	-.338	.487	.237	.763	1.101	.768	-.433	-.356	-.0027
.0125	.906	-.094	-.075	.925	.365	1.290	1.665	-.665	.560	.314	.686	1.351	.685	-.320	-.296	-.0037
.0250	1.039	.039	.031	1.031	.395	1.426	2.030	-1.030	.636	.404	.596	1.626	.540	-.145	-.150	-.0038
.0500	1.130	.130	.104	1.104	.394	1.498	2.240	-1.240	.710	.504	.496	1.736	.412	-.018	-.020	-.0010
.0750	1.176	.176	.141	1.141	.389	1.530	2.340	-1.340	.752	.565	.435	1.775	.347	.042	.048	.0036
.1000	1.207	.207	.166	1.166	.380	1.546	2.390	-1.390	.786	.618	.382	1.772	.303	.077	.090	.0090
.1500	1.245	.245	.196	1.196	.356	1.552	2.410	-1.410	.840	.706	.294	1.704	.246	.110	.132	.0198
.2000	1.270	.270	.216	1.216	.323	1.539	2.362	-1.362	.893	.798	.202	1.564	.211	.112	.136	.0272
.2500	1.288	.288	.231	1.231	.280	1.511	2.280	-1.280	.951	.904	.096	1.376	.185	.095	.117	.0292
.3000	1.300	.300	.240	1.240	.234	1.474	2.175	-1.175	1.006	1.012	-.012	1.163	.164	.070	.089	.0261
.3500	1.277	.277	.222	1.222	.188	1.410	1.989	-.989	1.034	1.070	-.070	.919	.140	.048	.059	.0207
.4000	1.252	.252	.202	1.202	.143	1.345	1.812	-.812	1.059	1.120	-.120	.692	.123	.020	.024	.0096
.4500	1.225	.225	.180	1.180	.103	1.283	1.645	-.645	1.077	1.160	-.160	.485	.108	-.005	-.006	-.0027
.5000	1.197	.197	.158	1.158	.069	1.227	1.507	-.507	1.089	1.185	-.185	.322	.095	-.026	-.030	-.0150
.5500	1.167	.167	.134	1.134	.042	1.176	1.382	-.382	1.092	1.191	-.191	.191	.082	-.040	-.045	-.0247
.6000	1.135	.135	.108	1.108	.020	1.128	1.271	-.271	1.088	1.183	-.183	.088	.072	-.052	-.058	-.0348
.6500	1.101	.101	.081	1.081	.004	1.085	1.177	-.177	1.077	1.160	-.160	.017	.062	-.058	-.063	-.0409
.7000	1.066	.066	.053	1.053	.002	1.055	1.115	-.115	1.051	1.105	-.105	.010	.054	-.052	-.055	-.0385
.7500	1.029	.029	.023	1.023	.003	1.026	1.050	-.050	1.020	1.040	-.040	.010	.045	-.042	-.043	-.0322
.8000	.989	-.011	-.009	.991	.004	.995	.990	.010	.987	.973	.027	.017	.038	-.034	-.034	-.0272
.8500	.947	-.053	-.042	.958	.005	.963	.928	.072	.953	.909	.091	.019	.029	-.024	-.023	-.0196
.9000	.901	-.099	-.079	.921	.004	.925	.856	.144	.917	.840	.160	.016	.023	-.019	-.017	-.0153
.9500	.853	-.147	-.118	.882	.002	.884	.782	.218	.880	.775	.225	.007	.014	-.012	-.010	-.0095
1.0000	.807	-.193	-.155	.845	0	.845	.714	.286	.845	.714	.286	0	0	0	0	0

$$\frac{V_t}{V_0} = \left(\frac{V}{V_0} - 1\right)\frac{t}{t_b} + 1$$

$$\frac{t}{t_b} = \frac{0.16}{0.20} = 0.80$$

Defendants' Exhibit A—(Continued)

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Consolidated Vultee Aircraft Corporation
San Diego Division

Model......Airplane Report No. ZA-101, App. B

Appendix B

Proposed Airfoil Ordinates & Profiles

Page 43 of 60

Consolidated Vultee Aircraft Corporation
San Diego Division

Model......Airplane Report No. ZA-101, App, B

Airfoil Ordinates

Airfoil and camber line ordinates were calculated by the method outlined in Reference 7. These calculations are summarized in Table I. Airfoil profiles calculated by the above procedure are given in Figures 1 and 2.

FIG 1

ROOT SECTION
63,4, 122
a = .1 b = .59

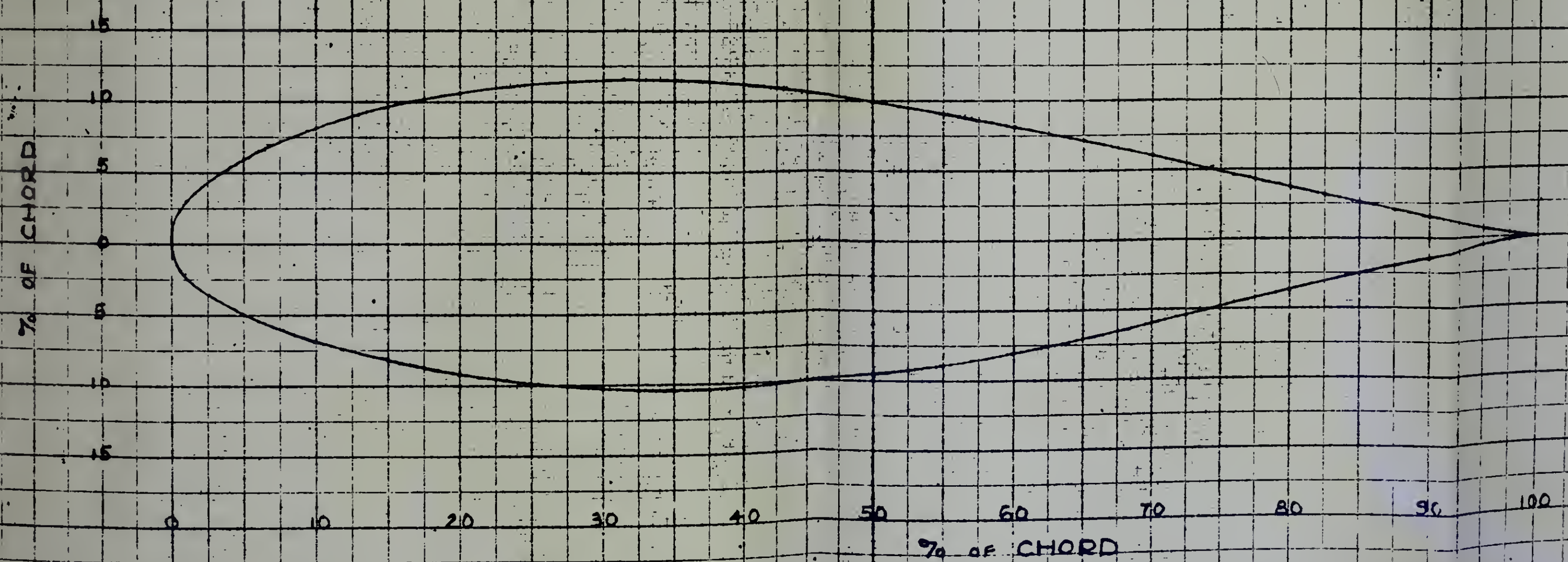
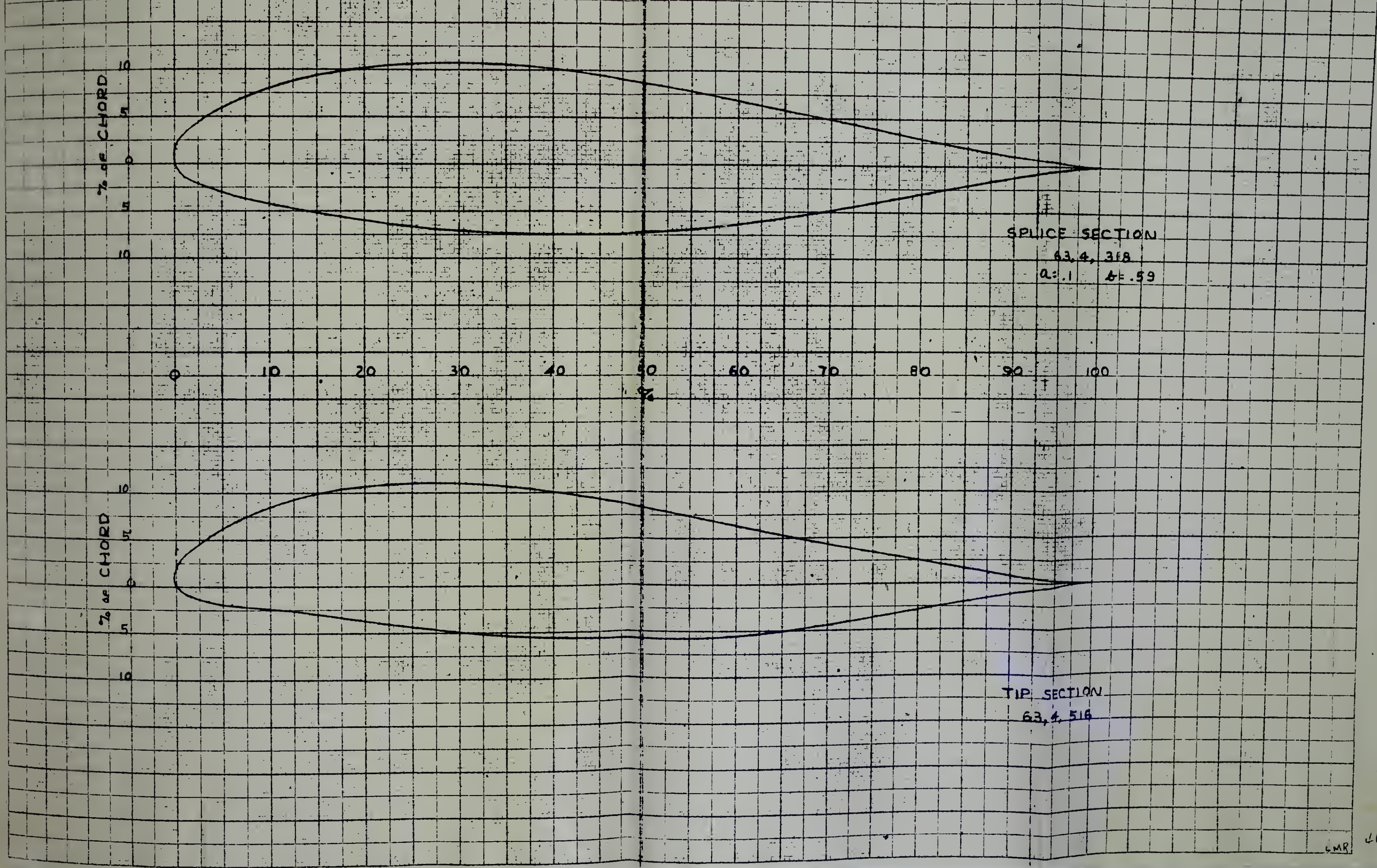


FIG 2


% of CHORD
10
5
0
5
10
SPLICE SECTION
63,4, 318
a=.1 b=.59
0
10
20
30
40
50
60
70
80
90
100
%
% of CHORD
10
5
0
5
10
TIP SECTION
63,4, 516

ORDINATES FOR PROPOSED AIRFOILS
REVISED WING

TABLE I

X	Root 63,4-122 (a=0.1, b=0.59) Yc	dyc/dx	Xu	Yu	Xl	Yl	0.48 Semi-Span 63,4-318 (a=0.1, b=0.59) Yc	dyc/dx	Xu	Yu	Xl	Yl	Tip 63,4-516 (a=0.1, b=0.59) Yc	dyc/dx	Xu	Yu	Xl	Yl
0	0		0	0	0	0	0		0	0	0	0	0		0	0	0	0
.005	.000579	.108701	.00296	.01953	.00704	-.01816	.001737	.326103	.00022	.01640	.00978	-.01293	.002895	.543505	-.00155	.01495	.01155	-.00915
.0075	.000919	.099130	.00524	.02370	.00976	-.02186	.002757	.297390	.00216	.02071	.01284	-.01519	.004595	.495650	.00011	.01952	.01489	-.01032
.0125	.001382	.086913	.00929	.03029	.01501	-.02753	.004146	.2[illegible]789	.00651	.02712	.01849	-.01883	.006910	.434565	.00409	.02626	.02091	-.01244
.025	.002350	.069777	.02224	.04186	.02776	-.03716	.007050	.21[illegible]31	.01805	.03875	.03195	-.02475	.011750	.348885	.01550	.03899	.03430	-.01549
.050	.003840	.051216	.04722	.05819	.05278	-.05051	.011520	.153[illegible]	.04324	.05553	.05676	-.03249	.019200	.256080	.04018	.05754	.05982	-.01914
.075	.004958	.038777	.07246	.07051	.07754	-.06059	.014874	.116[illegible]51	.06885	.06819	.08115	-.03644	.024790	.193885	.06592	.07163	.08408	-.02205
.10	.005788	.027386	.09795	.08036	.10205	-.06898	.017364	.0[illegible]58	.09499	.07816	.10501	-.04343	.028940	.136930	.09262	.08284	.10738	-.02496
.15	.006711	.010432	.14907	.09570	.15093	-.08228	.020133	.031296	.14772	.09291	.15228	-.05264	.033555	.052100	.14658	.09819	.15342	-.03107
.20	.006925	-.000273	.20003	.10600	.19997	-.09214	.020775	-.00[illegible]19	.20007	.10183	.19993	-.06028	.034425	-.001365	.20010	.10668	.19990	-.03742
.25	.006759	-.007957	.25084	.11269	.24916	-.09917	.019905	-.02[illegible]71	.25207	.10653	.24793	-.06674	.033795	-.039785	.25306	.11078	.24694	-.04318
.30	.006184	-.013459	.30147	.11566	.29853	-.10334	.018492	-.040377	.30362	.10801	.29639	-.07103	.030820	-.067255	.30535	.11028	.29465	-.04864
.35	.005381	-.017493	.35192	.11512	.34808	-.10436	.016143	-.052479	.35471	.10582	.34529	-.07354	.026905	-.087465	.35965	.10643	.34305	-.05261
.40	.004439	-.019992	.40215	.11184	.39785	-.10296	.013314	-.05[illegible]976	.40526	.10108	.39474	-.07423	.022195	-.099960	.40777	.09993	.39223	-.05554
.45	.003405	-.021106	.45217	.10842	.44783	-.09960	.010215	-.063518	.45533	.09434	.44467	-.07391	.017025	-.105530	.45786	.09155	.44214	-.05749
.50	.002353	-.020737	.50201	.09934	.49799	-.09464	.007059	-.062211	.50492	.08628	.49508	-.07216	.011765	-.103685	.50728	.08194	.49272	-.05840
.55	.001363	-.018457	.55165	.09091	.54835	-.08819	.004089	-.05[illegible]71	.55332	.07730	.54668	-.06912	.006815	-.092265	.55599	.071675	.54401	-.05604
.60	.000578	-.011635	.60004	.08143	.59906	-.08027	.001734	-.034905	.60231	.06785	.59769	-.06439	.002890	-.058175	.60342	.06160	.59658	-.05582
.65	.000130	-.006866	.65040	.07123	.64951	-.07097	.000390	-.026598	.65120	.05855	.64880	-.05777	.000650	-.034330	.65177	.05233	.64823	-.05103
.70	-.000138	-.004042	.70024	.06032	.69976	-.06060	-.000414	-.012126	.70060	.04905	.69940	-.04987	-.000690	-.020210	.70089	.04237	.69911	-.04465
.75	-.000287	-.001921	.75009	.04884	.74991	-.04942	-.000861	-.005765	.75023	.03933	.74977	-.04105	-.001435	-.009605	.75034	.03429	.74966	-.03717
.80	-.000347	-.000454	.80002	.03706	.79999	-.03776	-.001041	-.001562	.80004	.02957	.79996	-.03165	-.001735	-.002270	.80006	.02547	.79994	-.02895
.85	-.000337	.000808	.84998	.02542	.85002	-.02610	-.001011	.002424	.84995	.02007	.85005	-.02209	-.001685	.004040	.84992	.01705	.85008	-.02043
.90	-.000270	.001847	.89997	.01546	.90003	-.01510	-.000810	.003541	.89993	.01132	.90007	-.01294	-.001350	.009235	.89990	.00943	.90010	-.01213
.95	-.000155	.002723	.94998	.00535	.95002	-.00567	-.000465	.006169	.94996	.00404	.95004	-.00497	-.000775	.013616	.94995	.00323	.95005	-.00479
1.00	0	.003473	1.00000	0	1.00000	0	0	.010119	1.00000	0	1.00000	0	0	.017365	1.00000	0	1.00000	0

y_c – CAMBER LINE ORDINATES

dy_c/dx – SLOPE OF CAMBER LINE

x_u, y_u – COÖRDINATES OF UPPER SURFACE POINT

x_l, y_l – COÖRDINATES OF LOWER SURFACE POINT

Defendants' Exhibit A—(Continued)

Consolidated Vultee Aircraft Corporation
San Diego Division

Model......Airplane Report No. ZA-101, App. C

Appendix C

Span-Load Distributions

SPAN-LOAD CALCULATIONS

Span Load calculations for elevator zero and elevator up 10° are given in Tables I to IV. A graphical estimation of the stalling characterisitcs of the revised wing for these elevator conditions is given in Figures 1 and 2.

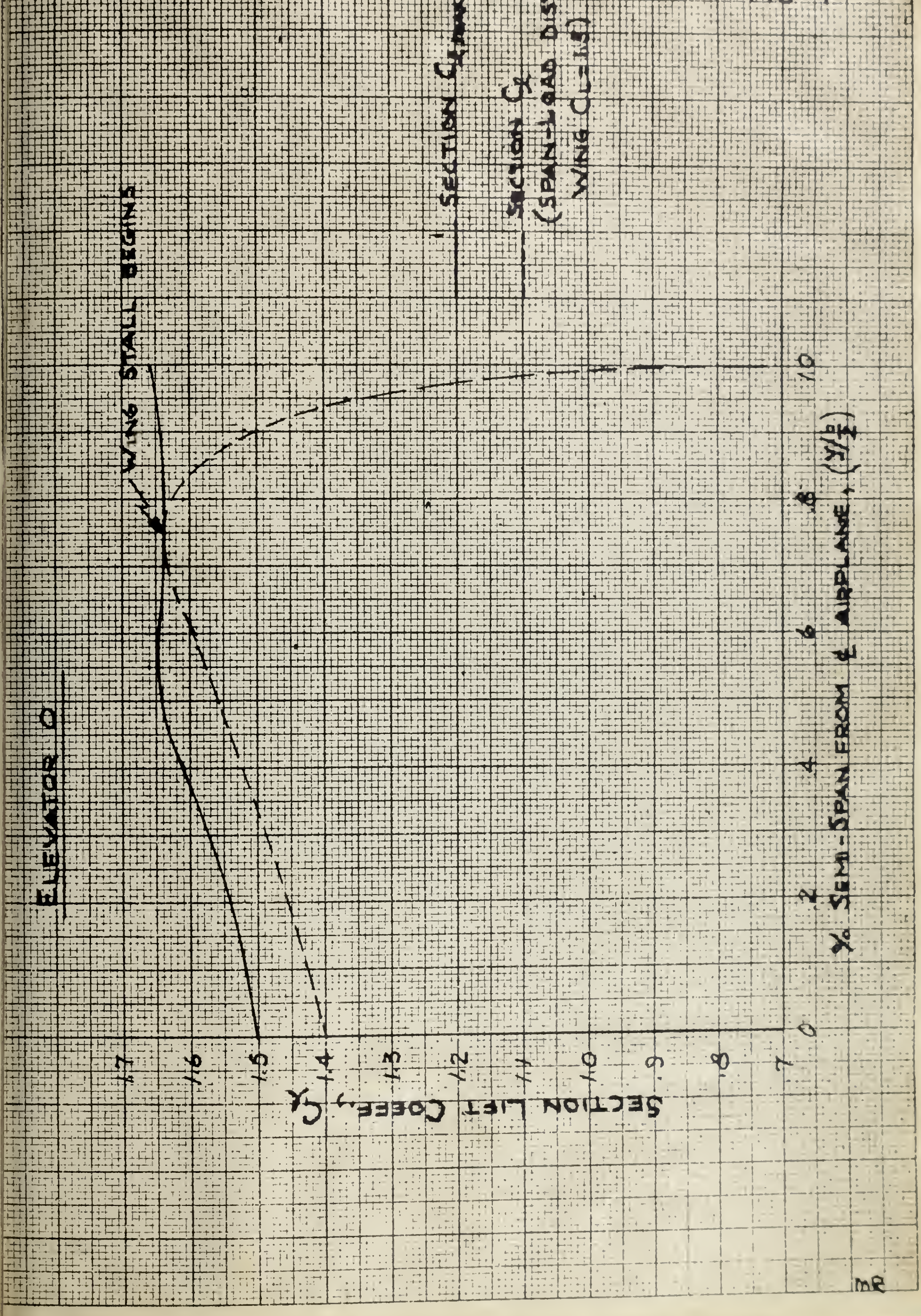
Pitching-moment calculations for the revised wing, elevator zero, were made by the method employed in ZA-056. For the estimation of full-scale results a correlation factor of 3 was applied to the calculated value given in Table II. As these calculations were made for a bare wing a ΔC_m of -.04 was added to these values for the estimation of complete model characteristics. Previous wind-tunnel tests of a tailless design show that this pitching-moment increment is a fair average for the change in pitching-moment due to the addition of fuselage, nacelles, etc.

BY

CHECKED

APPROVED

FIG 1


ELEVATOR 0
WING STALL BEGINS
SECTION $C_{l_{max}}$
SECTION C_l
(SPAN-LOAD DIST
WING C_L=1.5)
% SEMI-SPAN FROM ℄ AIRPLANE, $(\frac{y}{b/2})$
0
.2
.4
.6
.8
1.0
SECTION LIFT COEFF, C_l
1.7
1.6
1.5
1.4
1.3
1.2
1.1
1.0
.9
.8
.7

mr

1059

FIG 2

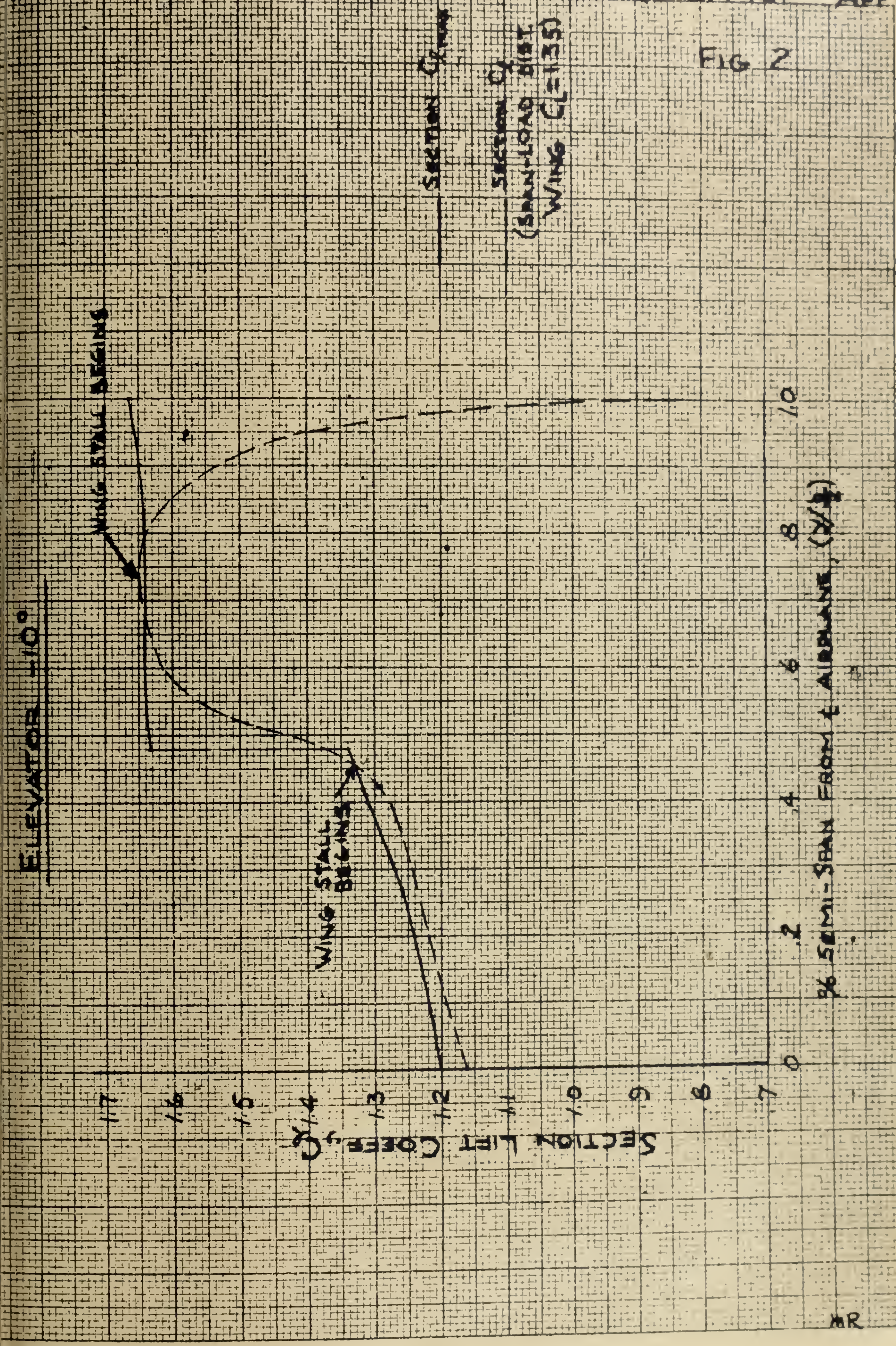
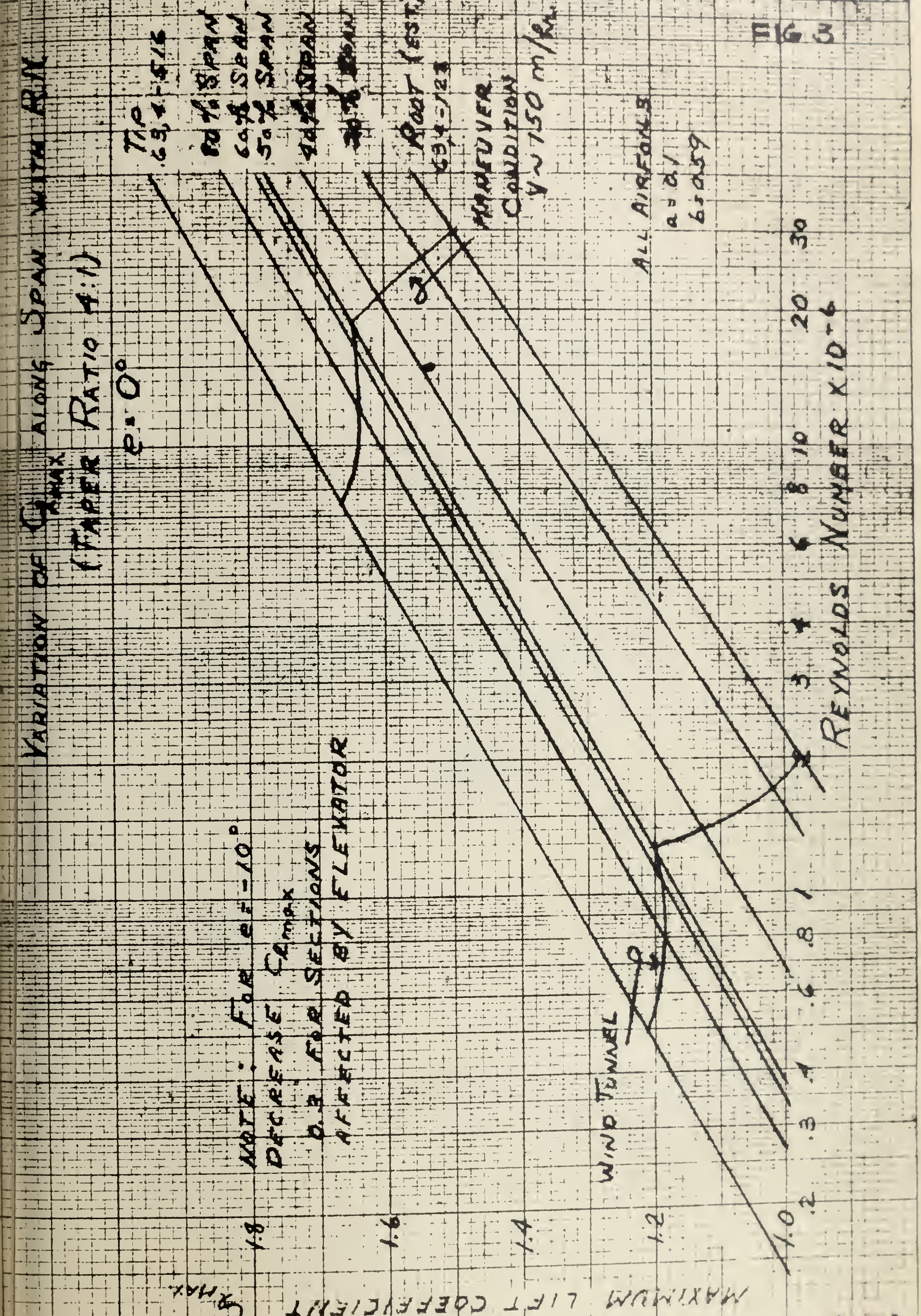

SECTION $C_{l_{max}}$
SECTION C_l (SPAN LOAD DIST. WING C_L=1.35)
WING STALL BEGINS
ELEVATOR -10°
WING STALL BEGINS
SECTION LIFT COEFF, C_l
% SEMI-SPAN FROM ℄ AIRPLANE, (y/b/2)
0
.2
.4
.6
.8
1.0
.7
.8
.9
1.0
1.1
1.2
1.3
1.4
1.5
1.6
1.7
MR

FIG 3


VARIATION OF $C_{L_{MAX}}$ ALONG SPAN WITH R.N.
(TAPER RATIO 4:1)
e: 0°
TIP
80% SPAN
60% SPAN
50% SPAN
40% SPAN
ROOT (EST)
MANEUVER CONDITION
V~150 m/hr
ALL AIRFOILS
NOTE: FOR e=-10° DECREASE $C_{L_{max}}$ 0.3 FOR SECTIONS AFFECTED BY ELEVATOR
WIND TUNNEL
1.8
1.6
1.4
1.2
1.0
.2
.3
.4
.6
.8
1
2
3
4
6
8
10
20
30
REYNOLDS NUMBER X10^-6
MAXIMUM LIFT COEFFICIENT $C_{L_{MAX}}$

SPAN-LOAD DISTRIBUTION : 2-ENG. TAILLESS DESIGN

REPORT ZA-101 APP. C

REVISED AIRFOILS & WING

ELEV. 0°

TABLE I

$$C_{\ell a_1} = \frac{1}{2}\left[\frac{a_0}{\bar{a}_0} + \frac{4\bar{c}}{\pi c}\sqrt{1-\left(\frac{y}{b/2}\right)^2}\right]$$

$$C_{\ell_b} = \frac{a_0}{2}\left(\alpha_{R_0} + \beta\right)$$

1	2	3	4	5	6	7	8	9	10	11	12	13	14	15	16	17	18	19	20
$y/\frac{b}{2}$	CHORD C (in)	ΔY in.	MULT. ×3	$\int c\,dy$ (2)×(3)×(4)	a_0 ($dC_L/d\alpha$)	$a_0/\bar{a}_0$	$4\bar{c}/\pi c$	$[y/\frac{b}{2}]^2$	$1-[y/\frac{b}{2}]^2$	$\sqrt{1-[y/\frac{b}{2}]^2}$ $\sqrt{(10)}$	$4\bar{c}/\pi c \times \sqrt{1-[y/\frac{b}{2}]^2}$ (8)×(11)	(7)+(12)	$C_{\ell a_1}$ (13)/2	ε WASHOUT (DEG.)	α_{ℓ_0}	β $(\varepsilon-\alpha_{\ell_0})$ (15)−(16)	$\int \beta c\,dy$ (17)×(5)	$\beta+\alpha_{R_0}$	C_{ℓ_b} $(a_0/2)\times(19)$
0	236.0	88.20	1	20800	.120	1.0	.794	0	1.000	1.000	.794	1.794	.897	0	-.200	.200	4160	.813	.049
.100	218.0	↓	4	77000	↓	↓	.860	.010	.990	.995	.855	1.855	.927	-.30	-.270	-.030	-2310	.583	.025
.200	200.0		2	35300			.938	.040	.960	.980	.918	1.918	.959	-.64	-.350	-.290	-10250	.223	.019
.300	183.0		4	64500			1.022	.090	.910	.954	.980	1.980	.990	-1.04	-.500	-.540	-34800	.073	.004
.400	165.0		2	29100			1.134	.160	.840	.915	1.042	2.042	1.021	-1.54	-.730	-.810	-23600	-.197	-.012
.500	147.0	↓	4	51800			1.274	.250	.750	.865	1.102	2.102	1.051	-2.04	-1.060	-.980	-50700	-.367	-.022
.600	129.0	88.20	1	11370			1.450	.360	.640	.800	1.160	2.160	1.080	-2.20	-1.170	-1.030	-11700	-.417	-.025
.600	129.0	44.10	1	5685			1.450	.360	.640	.800	1.160	2.160	1.080	-2.20	-1.170	-1.030	-5850	-.417	-.025
.650	120.0	↓	4	21200			1.560	.420	.580	.760	1.186	2.186	1.093	-2.30	-1.250	-1.050	-22300	-.437	-.026
.700	111.0		2	9780			1.690	.490	.510	.714	1.210	2.210	1.105	-2.42	-1.340	-1.080	-10550	-.467	-.028
.750	102.0		4	18000			1.840	.560	.440	.663	1.220	2.220	1.110	-2.56	-1.450	-1.110	-19980	-.497	-.030
.800	94.0		2	8290			1.990	.640	.360	.600	1.190	2.190	1.095	-2.74	-1.600	-1.140	-9450	-.527	-.032
.850	85.0	↓	4	15000			2.200	.720	.280	.529	1.162	2.162	1.081	-2.95	-1.780	-1.170	-17550	-.557	-.034
.900	76.0	44.10	1	3350			2.460	.810	.190	.436	1.070	2.070	1.035	-3.20	-2.060	-1.140	-3820	-.527	-.032
.900	76.0	22.05	1	1675			2.460	.810	.190	.436	1.070	2.070	1.035	-3.20	-2.060	-1.140	-1910	-.527	-.032
.925	71.5	↓	4	6310			2.620	.855	.145	.381	1.000	2.000	1.000	-3.35	-2.225	-1.125	-7100	-.512	-.031
.950	67.0		2	2960			2.800	.905	.095	.308	.863	1.863	.931	-3.55	-2.420	-1.120	-3320	-.507	-.031
.975	62.5	↓	4	5510			3.000	.950	.050	.224	.672	1.672	.836	-3.76	-2.650	-1.111	-6110	-.497	-.030
1.000	58.0	22.05	1	1280	↓	↓	3.230	1.000	0	0	0	1.000	.500	-4.00	-2.980	-1.020	-1305	-.407	-.025
				Σ=388910													4160 -242605 Σ-238445		

$$S = \frac{2}{3}\times 388910/144 = 1800\ \square'$$

$$\alpha_{R_0} = -\frac{1}{\left(\frac{b}{2}\right)\bar{c}}\int_0^{\frac{b}{2}} \beta c\,dy = -\left(\frac{-238445}{388{,}910}\right) = +.613^\circ$$

$$\bar{c} = \frac{(S/2)}{\left(\frac{b}{2}\right)} = \frac{1}{3}\times\frac{388910}{882} = 147 \text{ in.}$$

$$\frac{b}{2} = 882 \text{ in.}$$

SPANWISE PITCHING-MOMENT DISTRIBUTION
REVISED AIRFOILS & WING
ELEV. 0°

1	21	22	23	24	25	26	27	28	29	30
$y/\frac{b}{2}$	$C_{m_{a.c.}}$	$C_{m_{a.c.}}C$	X	$C_{l_b}X$	$C_{l_b}X+C_{m_{a.c.}}C$	$\int(25)c\,dy \times 10^{-5}$	C_{l_1} $(C_{l_{a_1}}+C_{l_b})$	$C_{l_1}X$	$C_{l_1}X+C_{m_{a.c.}}C$	$\int(29)c\,dy \times 10^{-5}$
		(21)×(2)		(20)×(23)	(22)+(24)	(25)×(5) ×10^-5	(20)+(14)	(23)×(27)	(22)+(28)	(29)×(5) ×10^-5
0	.0065	1.535	52.190	2.560	4.095	.850	.945	49.40	50.935	10.600
.100	.0083	1.810	39.110	1.370	3.180	2.450	.962	37.60	39.410	30.400
.200	.0105	2.110	26.020	.495	2.595	.918	.978	25.40	27.500	9.720
.300	.0120	2.380	12.940	.052	2.432	1.568	.994	12.90	15.280	9.850
.400	.0162	2.670	-.119	.001	2.671	.779	1.009	-.12	2.550	.743
.500	.0177	2.900	-12.196	.290	3.190	1.654	1.029	-13.50	-10.600	-5.500
.600	.0208	2.680	-26.270	.656	3.336	.379	1.055	-27.55	-24.870	-2.825
.600	.0208	2.680	-26.270	.656	3.336	.189	1.055	-27.55	-24.870	-1.413
.650	.0215	2.580	-32.810	.854	3.434	.728	1.067	-35.00	-32.420	-6.880
.700	.0225	2.480	-39.351	1.102	3.582	.350	1.077	-42.30	-39.820	-3.900
.750	.0230	2.345	-45.890	1.375	3.720	.670	1.080	-49.60	-47.255	-8.500
.800	.0242	2.280	-52.430	1.680	3.960	.328	1.063	-55.70	-53.420	-4.420
.850	.0255	2.165	-58.970	2.000	4.165	.625	1.047	-61.75	-59.585	-8.940
.900	.0273	2.075	-65.510	2.100	4.175	.140	1.003	-65.75	-63.675	-2.230
.900	.0273	2.075	-65.510	2.100	4.175	.070	1.003	-65.75	-63.675	-1.070
.925	.0282	2.020	-68.775	2.130	4.150	.262	.969	-66.50	-64.480	-4.060
.950	.0294	1.970	-72.044	2.230	4.200	.124	.900	-64.80	-62.830	-1.860
.975	.0308	1.925	-75.314	2.260	4.185	.231	.806	-60.65	-58.725	-3.240
1.000	.0325	1.885	-78.580	1.960	3.845	.049	.475	-37.40	-35.515	-.454
						Σ12.364				+61.313
										-55.292
										Σ+6.021

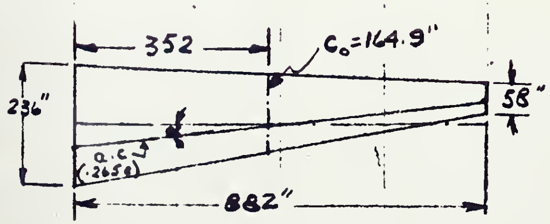
$$C_{m_{a.c.}} = \frac{\int (C_{l_1}x + C_{m_{a.c.}}C)\,c\,dy}{(\frac{S}{2})\,C_0} \quad (\text{FOR } C_L = 1.0)$$

$$C_{m_{a.c.}} = \frac{\int (C_{l_b}x + C_{m_{a.c.}}C)\,c\,dy}{(S/2)\,C_0} \quad (\text{FOR } C_L = 0)$$

$S/2\ C_0 = (5) \times 164.9 = 640.5$

$$C_{m_{ac}} = \frac{12.364}{640.5} = +.0193 \quad @\ C_L = 0$$

$$C_{m_{ac}} = \frac{6.021}{640.5} = +.0094 \quad @\ C_L = 1.0$$



$x = (352 - y)\tan\alpha$

$\tan\alpha = .14827$

SPAN-LOAD DISTRIBUTION 2-ENG. TAILLESS DESIGN
REVISED AIRFOILS & WING
ELEV. -10°

1 $y/\frac{b}{2}$	2 MULT. × 3	3 CHORD C (in.)	4 ΔY in.	5 $\int c\,dy$ (2)×(3)×(4)	6 ε WASHOUT (DEG.)	7 α_{ℓ_0}	8 β (ε − α_{ℓ_0}) (6)−(7)	9 $\int \beta c\,dy$ (8)×(5)	10 $\beta + \alpha_{R_0}$	11 C_{ℓ_b} (faired)
0	1	236	44.1	10400	0	4.40	−4.40	−45200	−.78	−.047
.05	4	227		40000	−.15	4.37	−4.52	−181000	−.90	−.054
.10	2	218		19250	−.30	4.33	−4.63	−89000	−1.01	−.060
.15	4	209		36900	−.45	4.30	−4.75	−175500	−1.13	−.069
.20	2	200		17650	−.64	4.25	−4.89	−86300	−1.29	−.077
.25	4	191		33700	−.83	4.18	−5.01	−169000	−1.39	−.083
.30	2	183		16150	−1.04	4.10	−5.14	−83000	−1.52	−.091
.35	4	174		30700	−1.26	4.00	−5.26	−161500	−1.64	−.098
.40	2	165		14580	−1.51	3.87	−5.41	−80400	−1.79	−.107
.45	4	156		27500	−1.83	3.70	−5.53	−152000	−1.91	−.090
.50	1	147		6480	−2.04	3.54	−5.58	−36150	−1.96	.0
.50	1	147		6480	−2.04	−1.06	−.98	−6350	+2.64	.0
.55	4	138		24300	−2.12	−1.10	−1.02	−24800	2.60	.125
.60	2	129		11400	−2.20	−1.17	−1.03	−11750	2.59	.155
.65	4	120		20200	−2.30	−1.25	−1.05	−21200	2.57	.154
.70	2	111		9800	−2.42	−1.34	−1.08	−10600	2.54	.152
.75	4	102		18000	−2.56	−1.45	−1.11	−20000	2.51	.150
.80	2	94		8300	−2.74	−1.60	−1.14	−9470	2.48	.149
.85	4	85		15000	−2.95	−1.78	−1.17	−17550	2.45	.147
.90	2	76		6720	−3.20	−2.06	−1.14	−7610	2.48	.149
.95	4	67		11800	−3.55	−2.43	−1.12	−13200	2.50	.150
1.00	1	58	↓	2560	−4.00	−2.98	−1.02	−2610	2.60	.154
				Σ 387870				Σ −1404850		

$$\alpha_{R_0} = -\left(\frac{-1404850}{387870}\right) = 3.62^\circ$$

ESTIMATION OF STALL CHARACTERISTICS
REVISED WING
2-ENG. TAILLESS DESIGN

TABLE IV

1 $y/\frac{b}{2}$	2 $C_{l_{a_1}}$	3 C_{l_b}	4 $C_{l_{a\,1.5}}$	5 $C_{L_{1.5}}$
		$e = 0°$		
0	.897	.049	1.348	1.397
.100	.927	.035	1.392	1.427
.200	.959	.019	1.440	1.459
.300	.990	.004	1.485	1.489
.400	1.021	-.012	1.533	1.521
.500	1.051	-.022	1.580	1.558
.600	1.080	-.025	1.620	1.595
.600	1.080	-.025	1.620	1.595
.650	1.093	-.026	1.643	1.617
.700	1.105	-.028	1.660	1.632
.750	1.110	-.030	1.665	1.635
.800	1.095	-.032	1.642	1.610
.850	1.081	-.034	1.625	1.591
.900	1.035	-.032	1.552	1.520
.900	1.035	-.032	1.552	1.520
.925	1.000	-.031	1.500	1.469
.950	.931	-.031	1.400	1.369
.975	.836	-.030	1.253	1.223
1.000	.500	-.025	.750	.725

6 $y/\frac{b}{2}$	7 C_{l_b} (faired)	8 $C_{l_{a\,1.35}}$	9 $C_{L_{1.35}}$
		$e = -10°$	
0	-.047	1.210	1.163
.10	-.060	1.253	1.193
.20	-.077	1.295	1.218
.30	-.091	1.335	1.244
.40	-.107	1.380	1.273
.50	0	1.420	1.420
.50	0	1.420	1.420
.55	.125	1.441	1.566
.60	.155	1.460	1.615
.65	.154	1.477	1.631
.70	.152	1.495	1.647
.75	.150	1.500	1.650
.80	.149	1.480	1.629
.85	.147	1.464	1.611
.90	.149	1.399	1.548
.95	.150	1.259	1.409
1.00	.154	.675	.829

Defendants' Exhibit A—(Continued)

Consolidated Vultee Aircraft Corporation
San Diego Division

Model......Airplane Report No. ZA-101, App. D

Appendix D

Method for the Calculation of the Leading Edge Radius of an Airfoil

ETHOD FOR THE CALCULATION OF THE LEADING EDGE RADIUS OF AN AIRFOIL

(Reference 8)

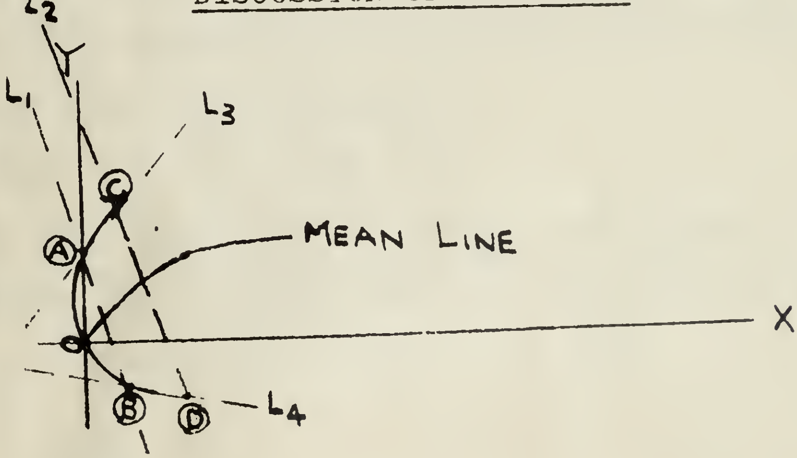
SUMMARY

The following method is used for the calculation of the
...g edge radius of an airfoil. The slope of the camber-line at
...on (0,0) shown in Appendix B is estimated in agreement with the
...of the camber line at station (0,0) as calculated by the pro-
...e outlined below.

Leading edge radii and mean camber-line slopes at station (0,0)
...he proposed revised wing airfoil section are:

	Radius	Slope
Root	3.7532 $\frac{c}{100}$	.1298
48% Span	2.3667 $\frac{c}{100}$	.3611
Tip	1.7159 $\frac{c}{100}$	.5961

DISCUSSION OF PROCEDURE



Select the four points (A) (B) (C) (D) which are nearest to
...e leading edge and of which the X and Y coordinates are known.

Then find the equation of the conic section through the four points and the origin. The radius of osculating circle at the origin is taken as the leading edge radius.

Notation m = slope

b = y intercept

$L = y - mx - b = 0$ is the equation of a straight line

Subscripts A, B, C, D, O refer to the points A, B, C, D, O.

Subscript 1 refers to a line AB passing through A & B.

Subscript 2 refers to a line CD.

Subscript 3 refers to a line AC.

Subscript 4 refers to a line BD.

Then (1) $m_1 = \dfrac{Y_A - Y_B}{X_A - X_B}$

(2) $b_1 = Y_A - m_1 X_A = Y_B - m_1 X_B$

Similarly m_2, m_3, m_4, b_2, b_3 & b_4 are found.

Using the degenerate form of the general conic as the sum of products of two linear equations.

(3) $(L_1) \times (L_2) + k\,(L_3) \times (L_4) = 0$

(4) $(y - m_1 x - b_1)(y - m_2 x - b_2) + k(y - m_3 x - b_3)(y - m_4 x - b_4) = 0$

Imposing the condition that the conic has to pass through point O ($x = 0$, $y = 0$) and solving for k gives

(5)
$$k = -\frac{b_1 b_2}{b_3 b_4}$$

Taking x-derivatives of (4) yields

(6)
$$(y - m_1 x - b_1)\left(\frac{dy}{dx} - m_2\right) + \left(\frac{dy}{dx} - m_1\right)(y - m_2 x - b_2)$$
$$+ k\left[(y - m_3 x - b_3)\left(\frac{dy}{dx} - m_4\right) + \left(\frac{dy}{dx} - m_3\right)(y - m_4 x - b_4)\right] = 0$$

Setting $x = 0$, $y = 0$ and solving for $\frac{dy}{dx}$ yields the slope of conic at the origin

$$m_0 = + \frac{b_1 m_2 + b_2 m_1 + k(b_3 m_4 + b_4 m_3)}{b_1 + b_2 + k(b_3 + b_4)}$$

Again taking x-derivatives of (6) yields

$$(y-m_1 x-b_1) \frac{d^2y}{dx^2} + \left(\frac{dy}{dx} - m_1\right)\left(\frac{dy}{dx} - m_2\right)$$

$$+ \left(\frac{dy}{dx} - m_1\right)\left(\frac{dy}{dx} - m_2\right) + \frac{d^2y}{dx^2}(y-m_2x-b_2)$$

$$+ k\left[(y-m_3x-b_3)\frac{d^2y}{dx^2} + 2\left(\frac{dy}{dx} - m_3\right)\left(\frac{dy}{dx} - m_4\right)\right.$$

$$\left. + \frac{d^2y}{dx^2}(y-m_4x - b_4)\right] = 0$$

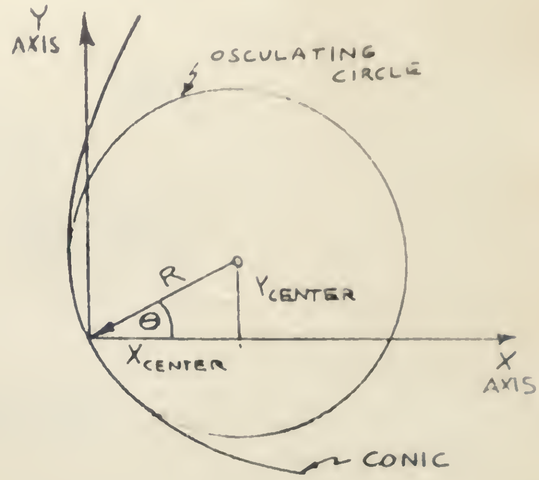
ing for $\frac{d^2y}{dx^2}$ and substituting $x = 0$; $y = 0$ gives

$$\frac{d^2y}{dx^2} = \frac{2\left[(1 + k)\, m_0^2 - \left\{m_1 + m_2 + k\,(m_3 + m_4)\right\} m_0 + m_1 m_2 + k m_3 m_4\right]}{b_1 + b_2 + k\,(b_3 + b_4)}$$

It is now possible to use the general equation for the radius he osculating circle for point 0

$$(10) \quad R_0 = \frac{\left[1 + m_0^2\right]^{3/2}}{\left(\frac{d^2y}{dx^2}\right)_0}$$

The coordinates of the center of the circle of radius R are



11 $X_{center} = R_0 \cos \Theta$

12 $Y_{center} = R_0 \sin \Theta$

The value of Θ may be found, where:

1) $\cot \Theta = -m_0$ or if no tables of functions are available

1) $\cos \Theta = -m_0 / \sqrt{1+m_0^2}$

1) $\sin \Theta = 1 / \sqrt{1+m_0^2}$

Admitted November 22, 1950.

BY

CHECKED

APPROVED

DEFENDANTS' EXHIBIT EE

The Glenn L. Martin Company — Model B-26
Baltimore, Maryland — Page No. 1

G.L.M. Engineering Report No. 1326

Wind Tunnel Investigation of the B-26
Stall Characteristics

Engineering Report No. 1326

The Glenn L. Martin Company
Baltimore, Maryland

July 19, 1940

Prepared by: A. J. Trimble, Jr.

Checked by: E. B. Schaefer.

Approved by: V. Outman,
Chief of Aerodynamics.

Approved by: Paul E. Hovgard,
Chief Research Engineer.

Defendants' Exhibit EE—(Continued)

The Glenn L. Martin Company — Model B-26
Baltimore, Maryland — Page No. 2

G. L. M. Eng. Rep. No. 1326

Contents

Defendants' Exhibit EE—(Continued)

The Glenn L. Martin Company Model B-26
Baltimore, Maryland Page No. 3

G.L.M. Eng. Rep. No. 1326

Wind Tunnel Investigation of B-26
Stall Characteristics

After a reconsideration of the probable stalling characteristics on the Model B-26 (Glenn L. Martin Model 179) it was decided that instead of waiting until the airplane is flown to see if tip stall occurs, a change should be made to diminish the possibility of poor behavior at the stall. An extensive test program was conducted at the Massachusetts Institute of Technology, Wright Brothers' Wind Tunnel, to determine the steps to be taken and the results are reported herein.

The scope of the investigation, and necessarily this report, was limited to those physical changes deemed advisable on the actual airplane in order not to delay delivery. Change in wing profile shape has been confined, therefore, to an area forward of the 10% chord line and outboard of Station 255 to the tip. In addition, the use of spoilers was also considered a possibility in the event that other methods failed to produce the correct effects.

With these limitations in mind, it appears that Leading Edge No. 2, illustrated on page 11 produces the desired effect most efficiently. In the following report, the justification of this choice will be brought forth by first, a short discussion of the basic

Defendants' Exhibit EE—(Continued)

problems of wing stall; second, a description of the model and tests; and third, a presentation and discussion of the data.

Basic Considerations

The criterion for desired stalling characteristics of an airplane must first be agreed upon, after which several methods for obtaining these characteristics are open to the designer along with methods for analyzing and predicting the results. In the

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case of the B-26, the field of possible wing design in this stage of the airplane's construction is limited because of a time consideration.

Desired Stalling Characteristics

The stall should start at the wing root to produce the most desirable and safest effect. Such a condition will result in a reduced downwash at the tail causing a diving moment tending to prevent the pilot from increasing the angle of attack and stalling the tips. Tail buffeting, a result of the turbulent air from the stalled root sections, warns the pilot that he has reached a stalled condition. A mid-panel stall between the nacelle and inboard end of the aileron causes neither serious tail buffeting nor a diving moment. The desirability of completely eliminating tip stall is universally recognized.

Defendants' Exhibit EE—(Continued)

Design Methods and Limitations

The desired stall may be regulated; first, by warping the wing either geometrically or aerodynamically; second, varying plan form shape, i.e., taper ratio; third, varying thickness ration along the span; fourth, using slots to delay the stall; fifth, using spoilers to cause stall.

The section of wing available for design change to assure root stall is illustrated on page 10. These limitations narrow the field of design methods to a leading edge change which might incorporate a slot, or a drooped nose effectively warping the tip of the wing. There is no design limitation on the use of spoilers.

In addition to physical limitations, further restrictions are present because the root stall must be obtained with the least possible increase in drag and the greatest possible increase in maximum lift.

The Glenn L. Martin Company — Model B-26
Baltimore, Maryland — Page No. 5

G.L.M. Eng. Rep. No. 1326

Methods of Stall Analysis

The stall of the B-26 wing has been analyzed both at full scale Reynold's Number and Model Reynold's Number in accordance with the method set forth in NACA Technical Report No. 572. The results appear on pages 12 and 13. Approximate values of $C_{L\,max}$ and ΔC_L 's due to varying Reynold's Number were estimated from the data available on

Defendants' Exhibit EE—(Continued)

the standard symmetrical airfoil series, (NACA OO—) with the maximum thickness at the 30% chord station. Since the Model B-26 wing contours are those of the NACA OO—64 airfoils with the maximum thickness at the 40% chord station, it is very likely that some discrepancy may exist in these stall diagrams.

The knowledge that the model did not stall exactly as indicated by the stall diagram, but nearer the tip, led to an investigation of the possibilities of correcting this condition, for the same discrepancy might exist on the full scale airplane. Such a condition would be aggravated by propeller wash. For these reasons, a study of the changes possible on the airplane and subsequently a complete test program of the various corrective possibilities has been undertaken.

Method of Test

The test program was conducted in the Massachusetts Institute of Technology Wright Brothers' Wind Tunnel with an ⅛ scale model of the B-29 which conformed in all respects to the airplane as being built.

Apparatus

The model used in this program was identical with the one used in previously reported tests (E.R. 1308). The leading edge of both wings was cut out as shown on page 10 to accommodate various

Defendants' Exhibit EE—(Continued)

The Glenn L. Martin Company — Model B-26
Baltimore, Maryland — Page 6

G.L.M. Eng. Rep. No. 1326

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leading edges which are illustrated on page 11. In addition the old B-26 model with the twisted wing (2° washout) was also tested. Flaps, airflow and Block Nacelles were available for the model.

The wind tunnel is equipped with a grid which raises the normal turbulence factor of 1.015 to 2.5. The speeds used were 125 M.P.H. with the grid and 150 M.P.H. without the grid. The accuracy obtainable in coefficient form for this model is:

$$C_D = .0002$$
$$C_L = .002$$

Procedure

To determine stall characteristics, pictures of tufts were taken at various angles of attack with the different leading edges at the same time lift data were taken. Tufts have no effect on lift in this particular case (Page 14). For these stall runs, the grid was used in the tunnel, and the model was equipped with airflow nacelles and deflected flaps (55°) because this configuration results in the most undesirable stall pattern, and is most likely to agree with flight conditions. The tail was not on the model during these runs because of the likelihood of severe buffeting. The airspeed was 125 M.P.H.

For drag tests, the model was equipped with block

Defendants' Exhibit EE—(Continued)

nacelles, flaps zero, and tail in place. No grid was used in the tunnel and the airspeed was 150 M.P.H.

In addition to these tests, unsymmetrical stalls were investigated.

Discussion

The lift and drag results and stall pictures are self-explanatory for the most part, but the choice of the best compromise is not quite as apparent from these data as it might be. The justification of the final choice is discussed in the following section.

Results

Pages 15 to 22 illustrate the stall patterns and lifts developed by the various leading edges. For the most part, the stalls were symmetrical. Occasionally a root stall occurred on the right side before the left wing had stalled. In the case of the No. 2 leading edge, as shown on page 16, the right wing stalled first at the aileron. The left wing could not be stalled. Close inspection showed slightly more camber in the left side than in the right. Premature stalling of one wing delays the stall of the other wing. By deliberately stalling the left wing with spoilers along the leading edge, the right wing was found to stall 1 to 1½ degrees later than with the left wing unstalled. Hence, an apparent difference in stall commencement of 2 degrees between

Defendants' Exhibit EE—(Continued)

the right and left wing may be brought about by a ½ degree discrepancy in wing contours or stream rotation. To correct for this condition, the right wing with the No. 2 leading edge was mudded to attempt to develop a symmetrical stall. The $C_{L\,max}$ obtained is shown on page 24 and is considered the best estimate of the performance of this design.

A drag summary plot appears on page 25. On page 26 the drag of spoilers used to produce a root stall with leading edges numbers 1 and 2 are plotted.

Stall Characteristics and $C_{L\,max}$

Inspection of this data indicates that no leading edge satisfies the requirement that the stall start at the root. Reference to the stall characteristics plots on pages 12 and 13 shows that at full scale Reynold's Number, there should be a tendency for the root to stall relatively sooner with respect to the tip than at model scale Reynold's Number. Leading edges Nos. 3, 6, and 7 satisfactorily delay tip stall but create a stall in the mid-panel, an unsatisfactory condition, and it very likely that scale effect will not be great enough to transfer this mid-

panel stall to the root. In other words, a leading edge design permitting the tip sections to reach too high a lift coefficient when the wing sections adjacent to this leading edge change are approaching

Defendants' Exhibit EE—(Continued)

a stall condition, will hasten the stall in these unchanged mid-panel sections. Too great a delay in tip stall must be avoided, or drastic spoiling of the root section lift will be required, resulting in a very low overall maximum lift coefficient. Verification of this fact is apparent in the maximum lift comparisons of page 23, where the No. 7 leading edge affords a lower overall $C_{L\,max}$ than the No. 2 leading edge.

From the stall standpoint, the No. 2 leading edge is the best solution. $C_{L\,max}$ is increased .22, (page 24) indicating a delayed tip stall which is verified by the stall pictures. Scale effect enhances the use of this leading edge because the stall characteristic plots of pages 12 and 13 show an increased margin at the tip and decreased margin at the root; at the same time, leading edge No. 2 will not effect a large enough change in lift distribution to bring about a premature stall in the mid-panel. Admittedly, a root stall is not produced on the model by the use of this design, but scale effect will tend to change the stall characteristics of the model, moving the stall inboard to the root.

If this condition is not realized in flight, it may be obtained by placing small spoilers on the root section similar to those tested on the model (page 10). The change in lift caused by the spoilers is plotted on page 24. The actual lift and stall pictures are shown on page 21. Should the spoilers be needed in flight, the resulting airplane characteristics will be more acceptable than with any other leading edge design, for, as already pointed out, the other leading edges would produce a very definite mid-panel stall

Defendants' Exhibit EE—(Continued)

which could necessitate large spoilers on the root section to move the stall inboard. The result would be a very marked decrease in $C_{L\,max}$ and a great increase in drag. A root stall may be produced on the original wing with spoilers but the resultant $C_{L\,max}$ is extremely low in comparison with leading edge No. 2 (page 24). Comparisons of the original twisted B-26 wing and the more recent wing show very little difference in stall characteristics or $C_{L\,max}$ (pages 15 and 22).

The Genn L. Martin Company | Model B-26
Baltimore, Maryland | Page No. 9

G.L.M. Eng. Rep. No. 1326

Drag

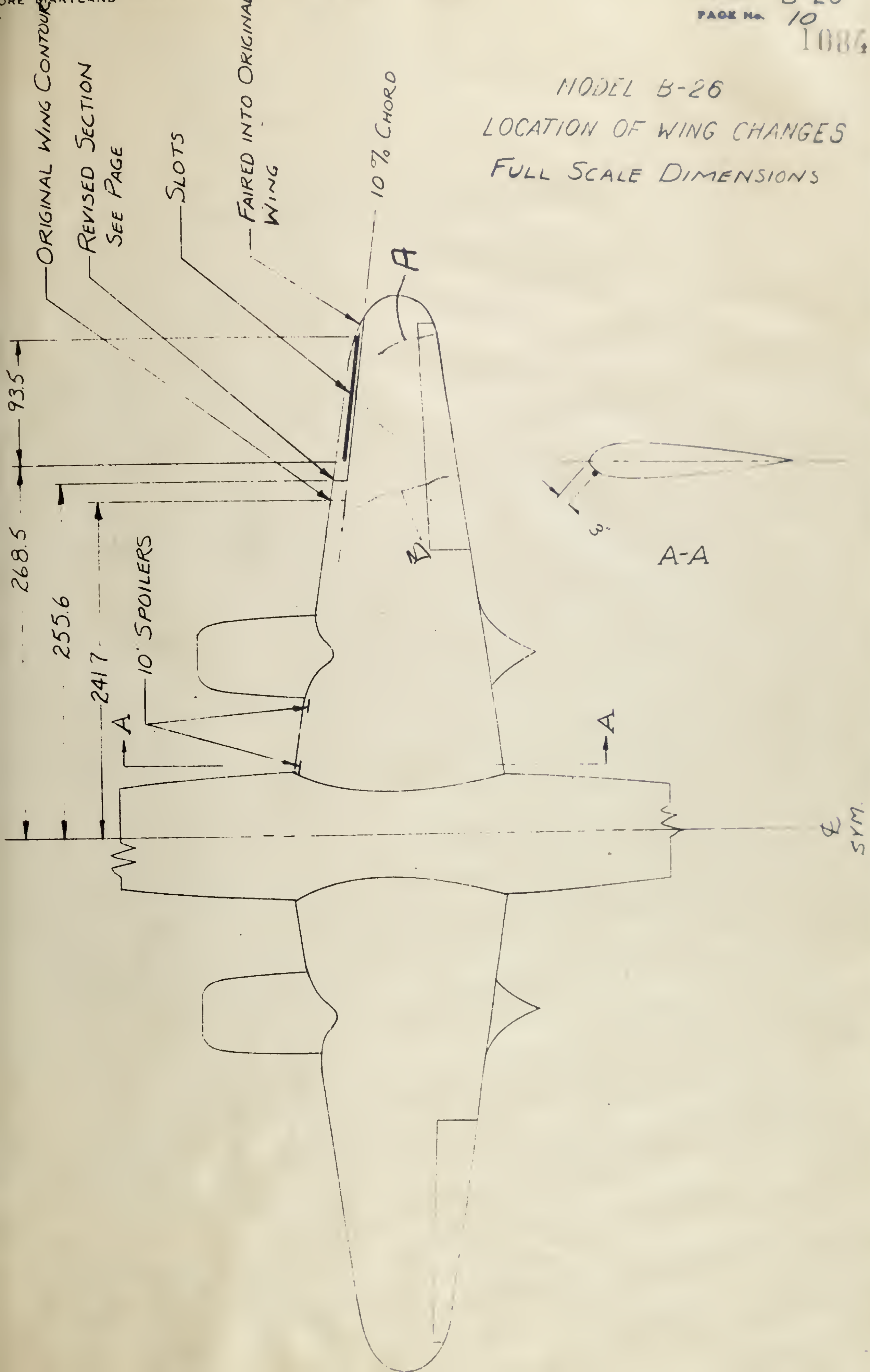
In addition to affording the best compromise in $C_{L\,max}$ and stall pattern, the No. 2 leading edge has less drag than any of the other configurations. Should the spoilers be found necessary, the drag is also slight. The comparisons of the No. 1 and No. 2 leading edge drags with spoilers is shown on page 26.

Conclusions

As a result of this investigation, the No. 2 leading edge is being incorporated in the design of the B-26 wing.

In case the stalling characteristics are not quite satisfactory in flight, it will be possible to completely correct it by adding a spoiler similar to those tested on the wind tunnel model.

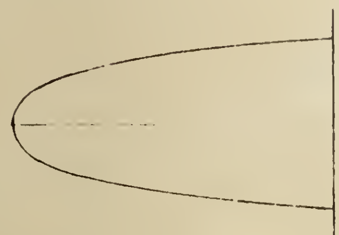
MODEL B-26
LOCATION OF WING CHANGES
FULL SCALE DIMENSIONS



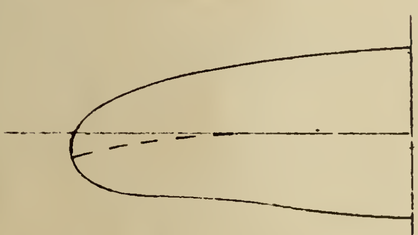
MODEL B-26

TYPICAL SECTIONS THROUGH LEADING EDGE AT STA. 350

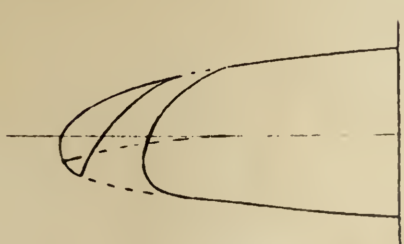
NOTE: ALL MODIFICATIONS FAIR INTO ORIGINAL CONTOUR AT 10 PERCENT CHORD STATION.



LEADING EDGE NO. 1
ORIGINAL CONTOUR
REF. DWG. W.T. 179-33

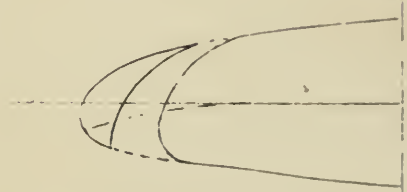


LEADING EDGE NO. 2
ORIGINAL CONTOUR MOD-
IFIED APPROX TO
220-- SERIES AIRFOIL
REF DWG W.T. 179-196 2 OF 7



LEADING EDGE NO. 3
SAME AS CONTOUR NO. 2
SLOT ADDED
REF. DWG W.T. 179-196 3 OF 7

LEADING EDGE NO. 5
SAME AS CONTOUR NO 1
SLOT ADDED
REF. DWG. W.T. 179-96 5 OF 7



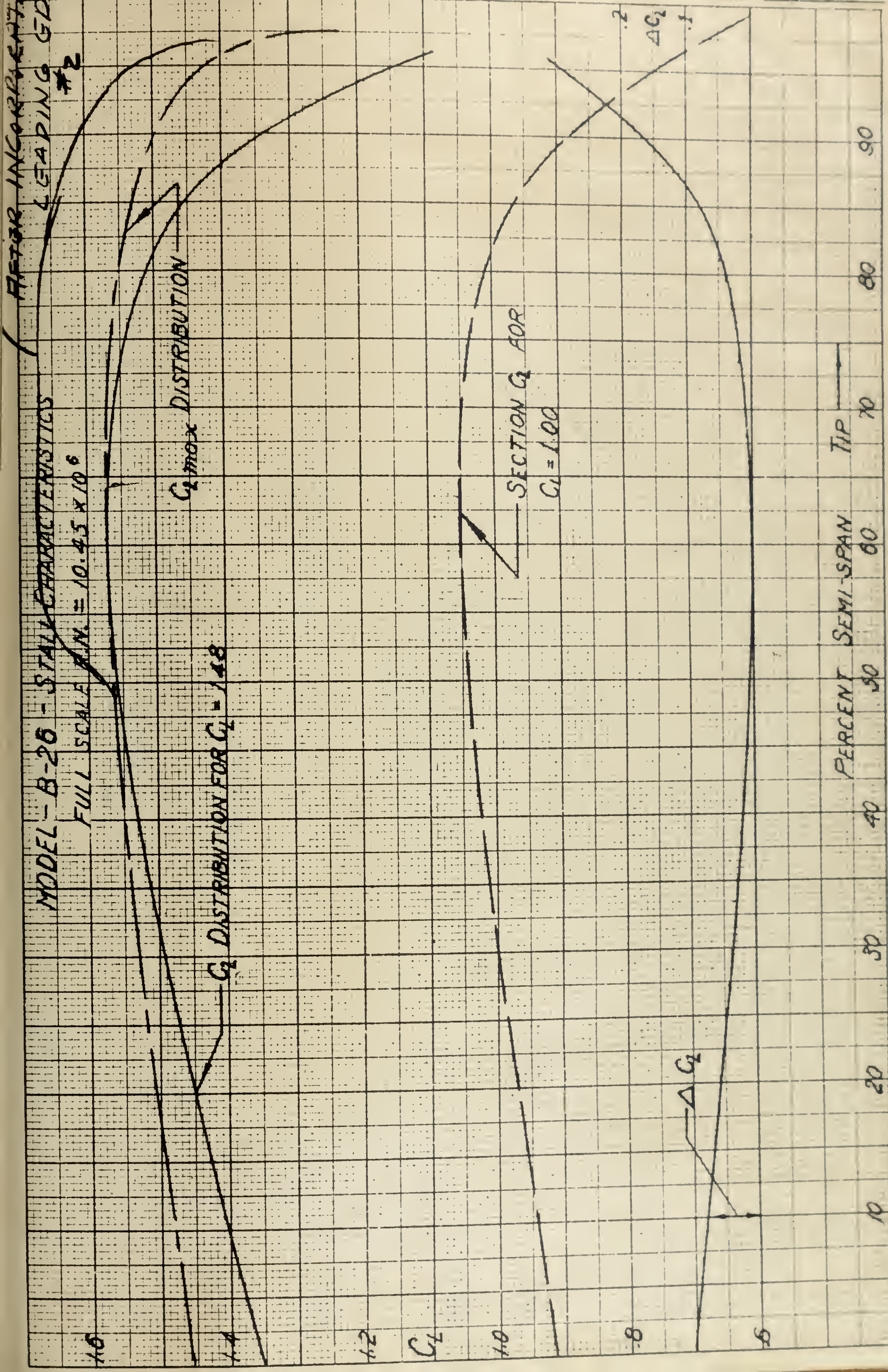
LEADING EDGE NO. 6
CONTOUR NO 2 WITH SLOT
REF DWG W.T. 179-196 6 OF 7

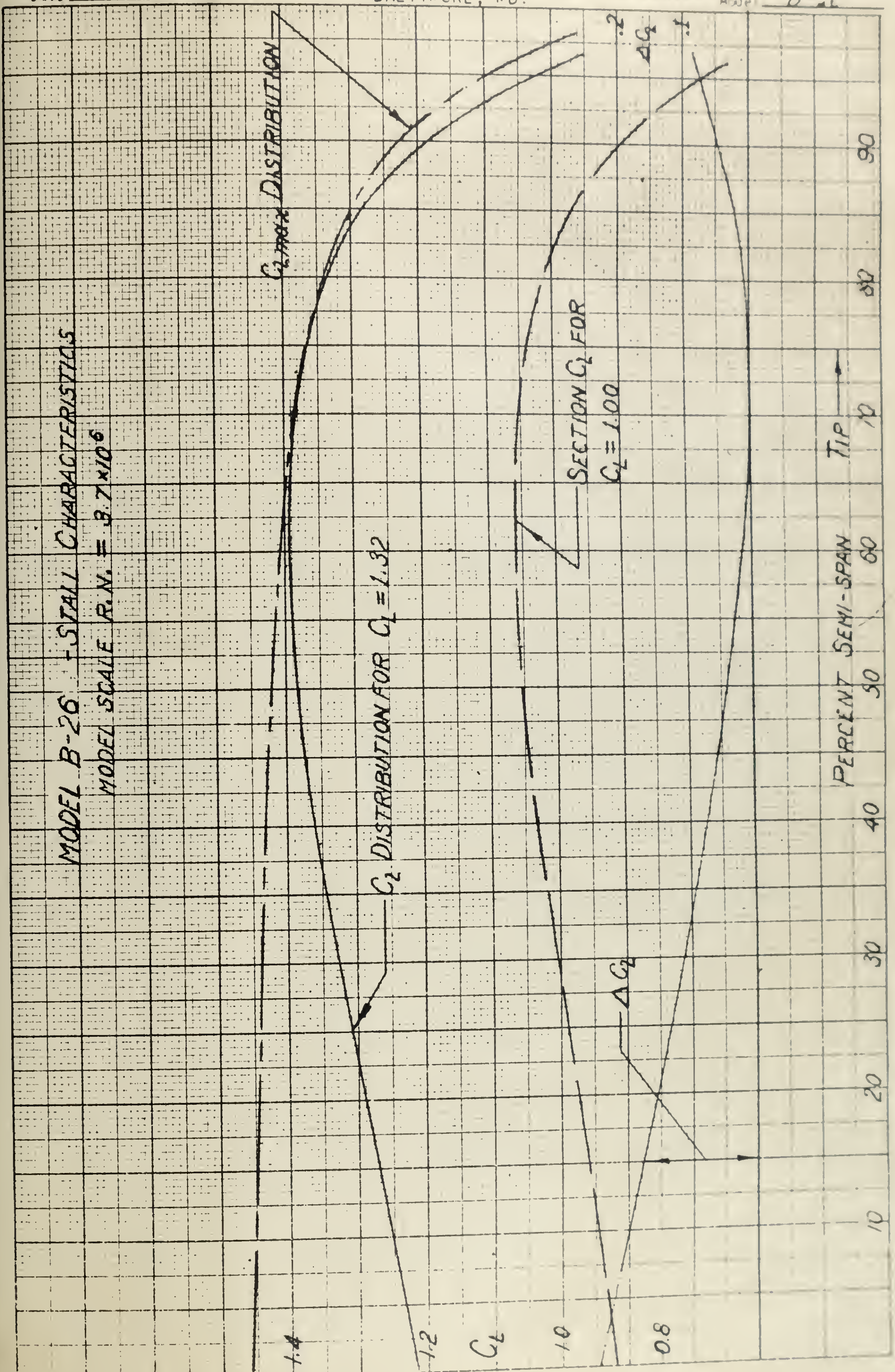
LEADING EDGE NO. 7
ORIGINAL CONTOUR MODIFIED APP.
TO 420-- SERIES AIRFOIL
REF DWG W.T. 179-196 7 OF 7

Prepared By ______ Date ______ 40

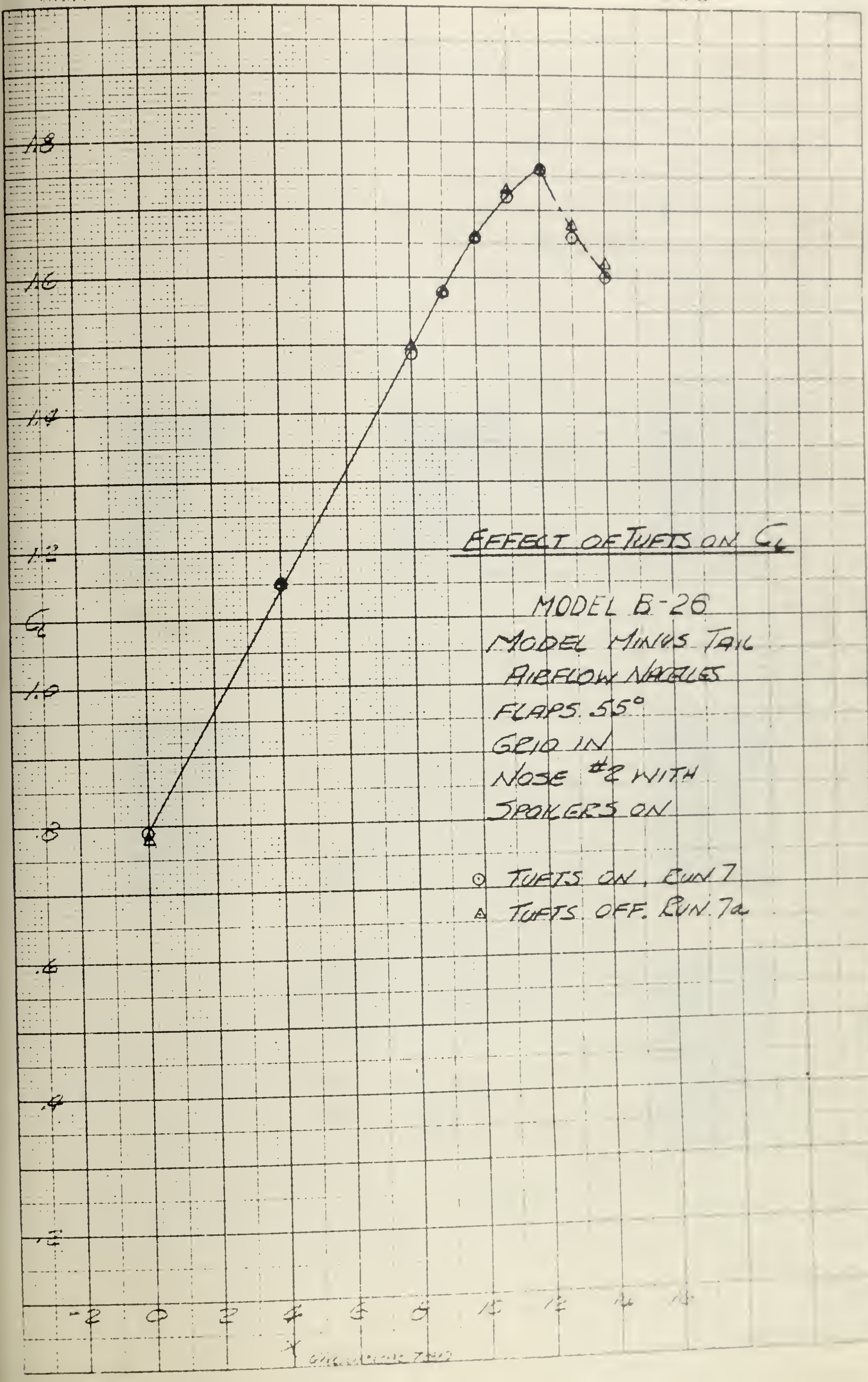
THE GLENN L. MARTIN COMPANY
BALTIMORE, MD.

Page 12 1086
Report No. 1326
Model B-26




MODEL B-26 - STALL CHARACTERISTICS
MODEL SCALE R.N. = 3.7×10⁶
$C_{l\,max}$ DISTRIBUTION
C_l DISTRIBUTION FOR $C_L = 1.32$
SECTION C_l FOR $C_L = 1.00$
ΔC_l
ΔC_l
.2
.1
1.4
1.2
C_L
1.0
0.8
10
20
30
40
50
60
70
80
90
PERCENT SEMI-SPAN
TIP

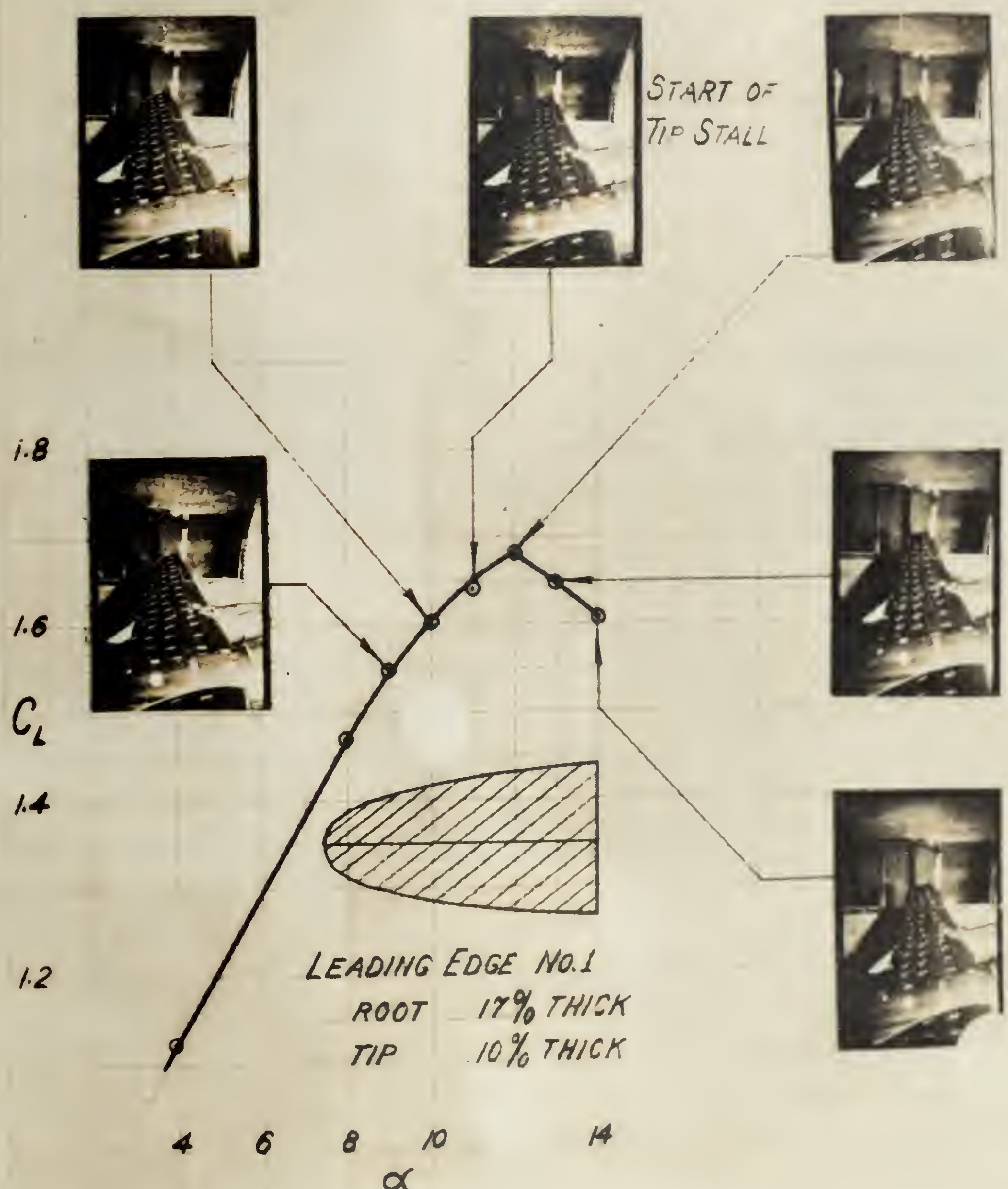
1088


EFFECT OF TUFTS ON C_L
MODEL B-26
MODEL MINUS TAIL
AIRFLOW NACELLES
FLAPS 55°
GRID IN
NOSE #2 WITH
SPOILERS ON
TUFTS ON, RUN 7
TUFTS OFF, RUN 7a
C_L
1.8
1.6
1.4
1.2
1.0
.8
.6
.4
.2
-2
0
2
4
6
8
10
12
14
16

MODEL 179

C_L VS α AND STALL PATTERNS

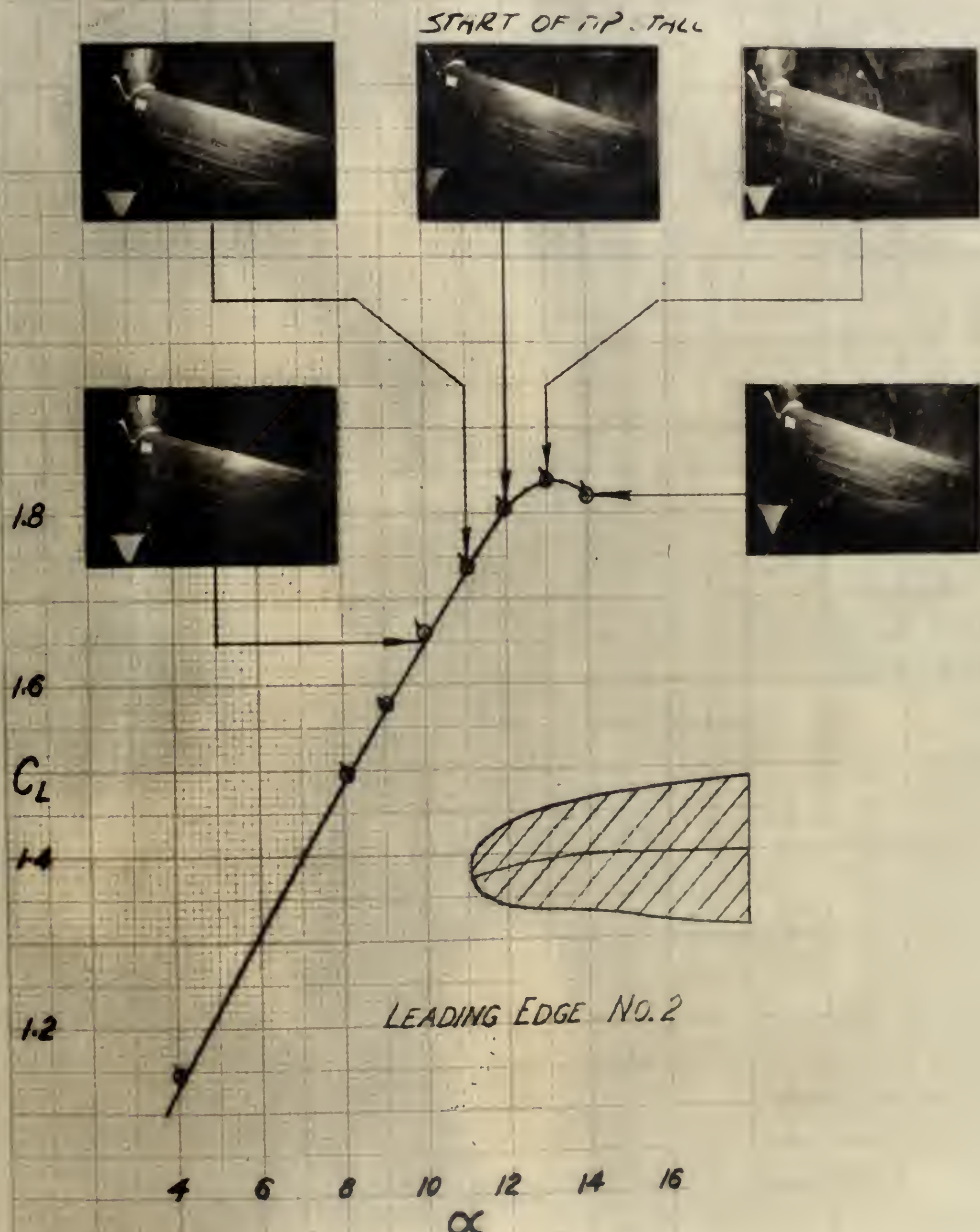
R.N. = 3.7×10^6
FLAPS 55°
AIRFLOW NACELLES



MODEL 179

C_L VS α AND STALL PATTERNS

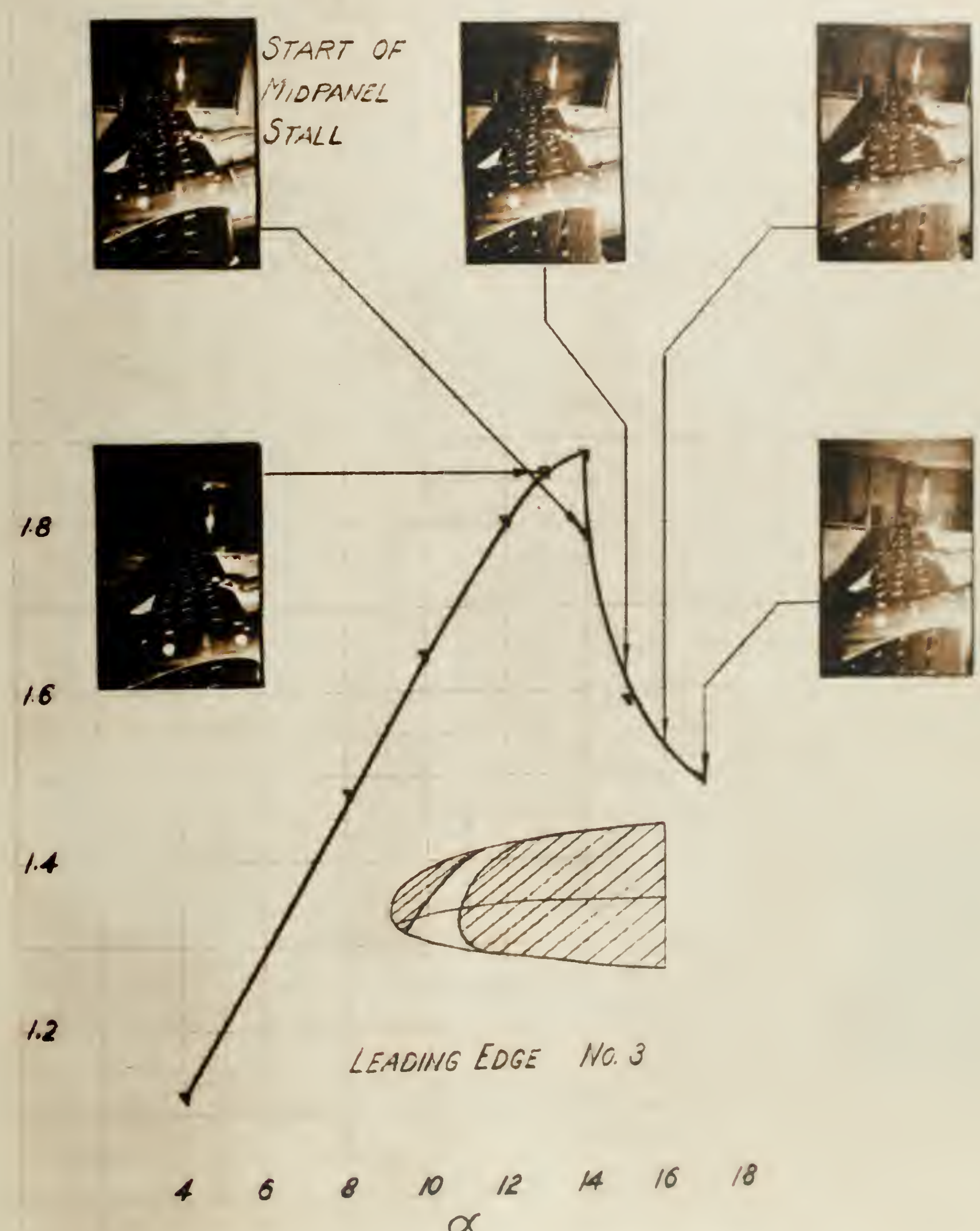
R.N. 3.7×10^6
FLAPS 55°
AIRFLOW NACELLES

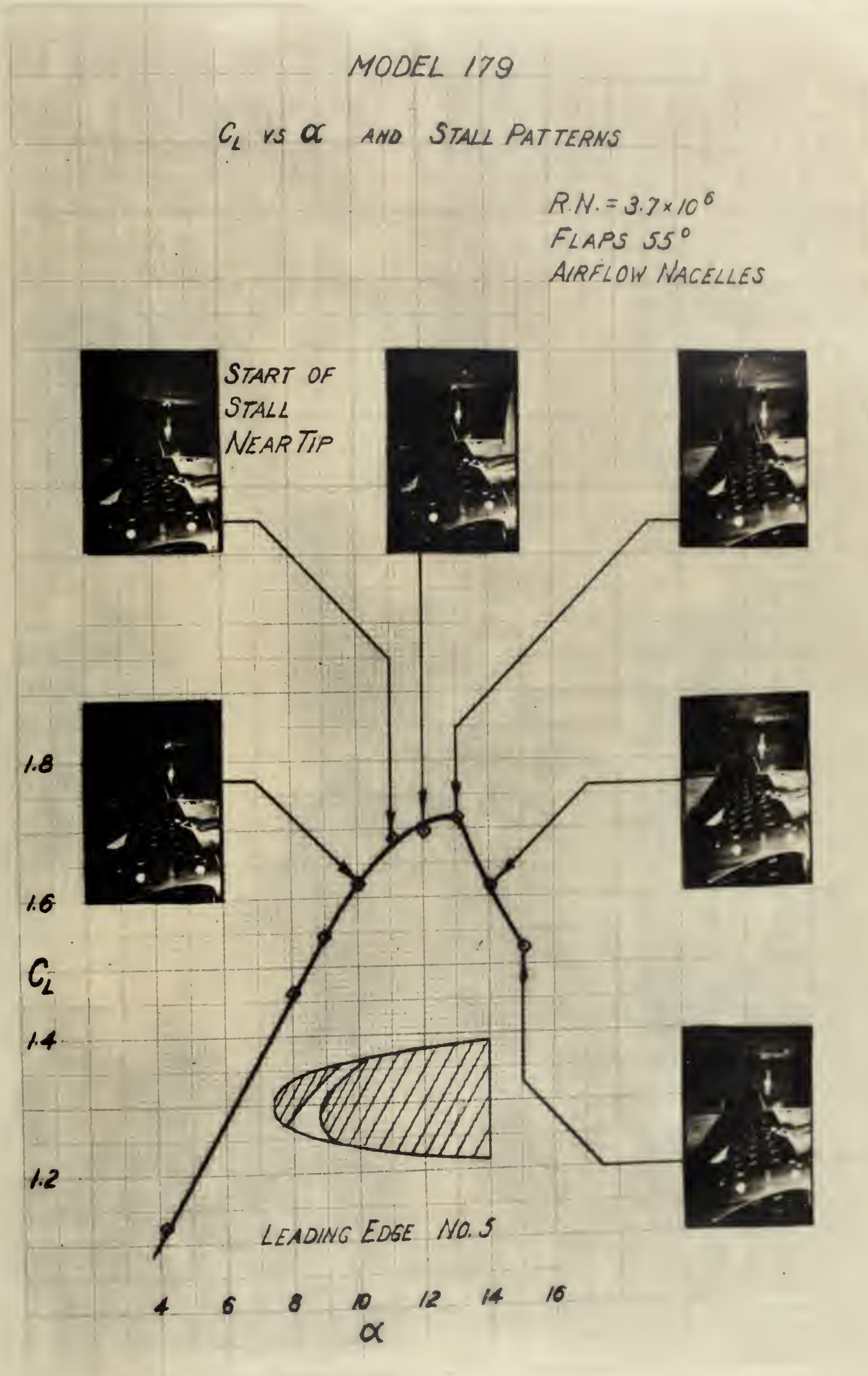


MODEL 179

C_L VS α AND STALL PATTERNS

R.N. = 3.7×10^6
FLAPS 55°
AIRFLOW NACELLES



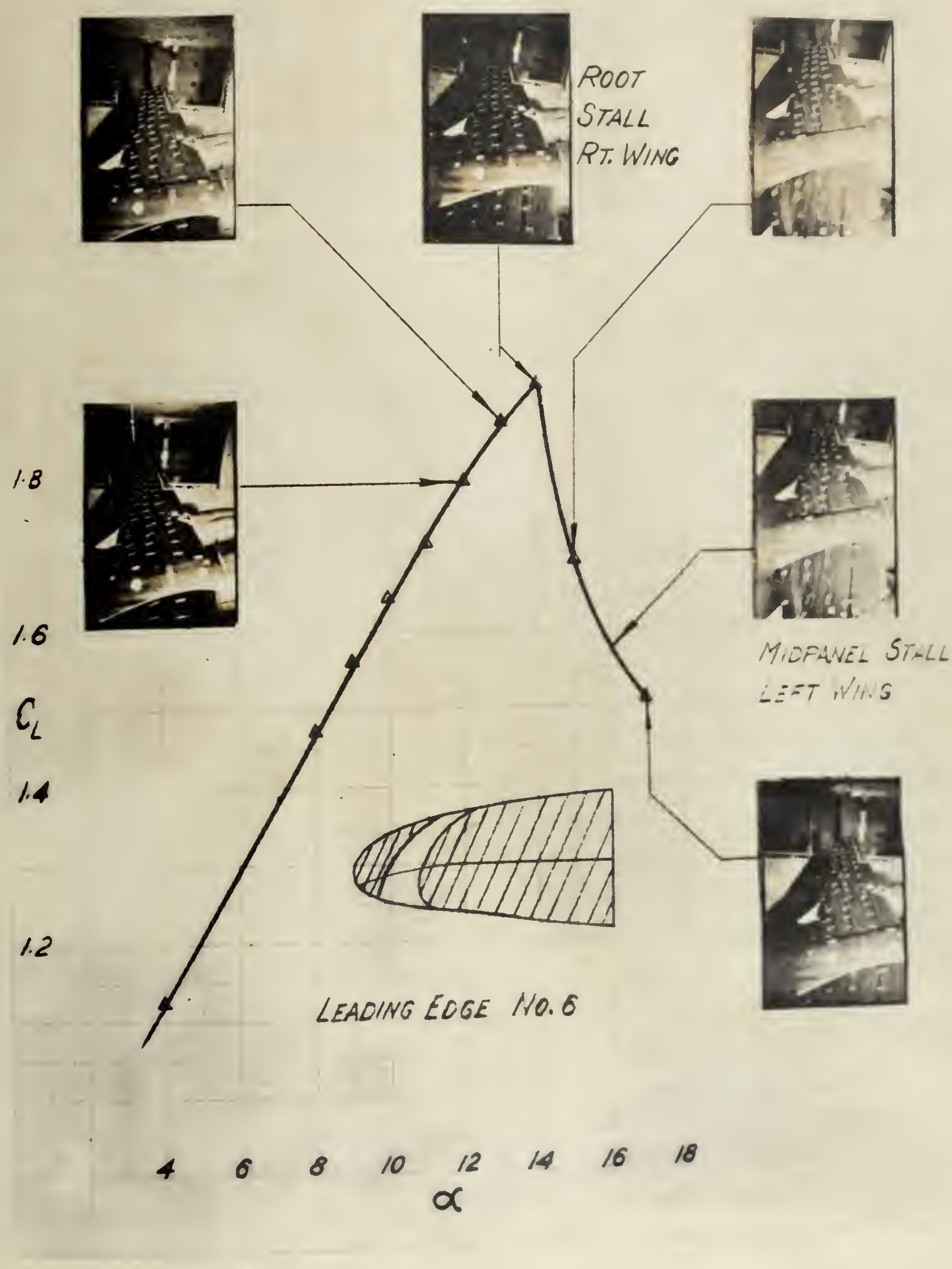


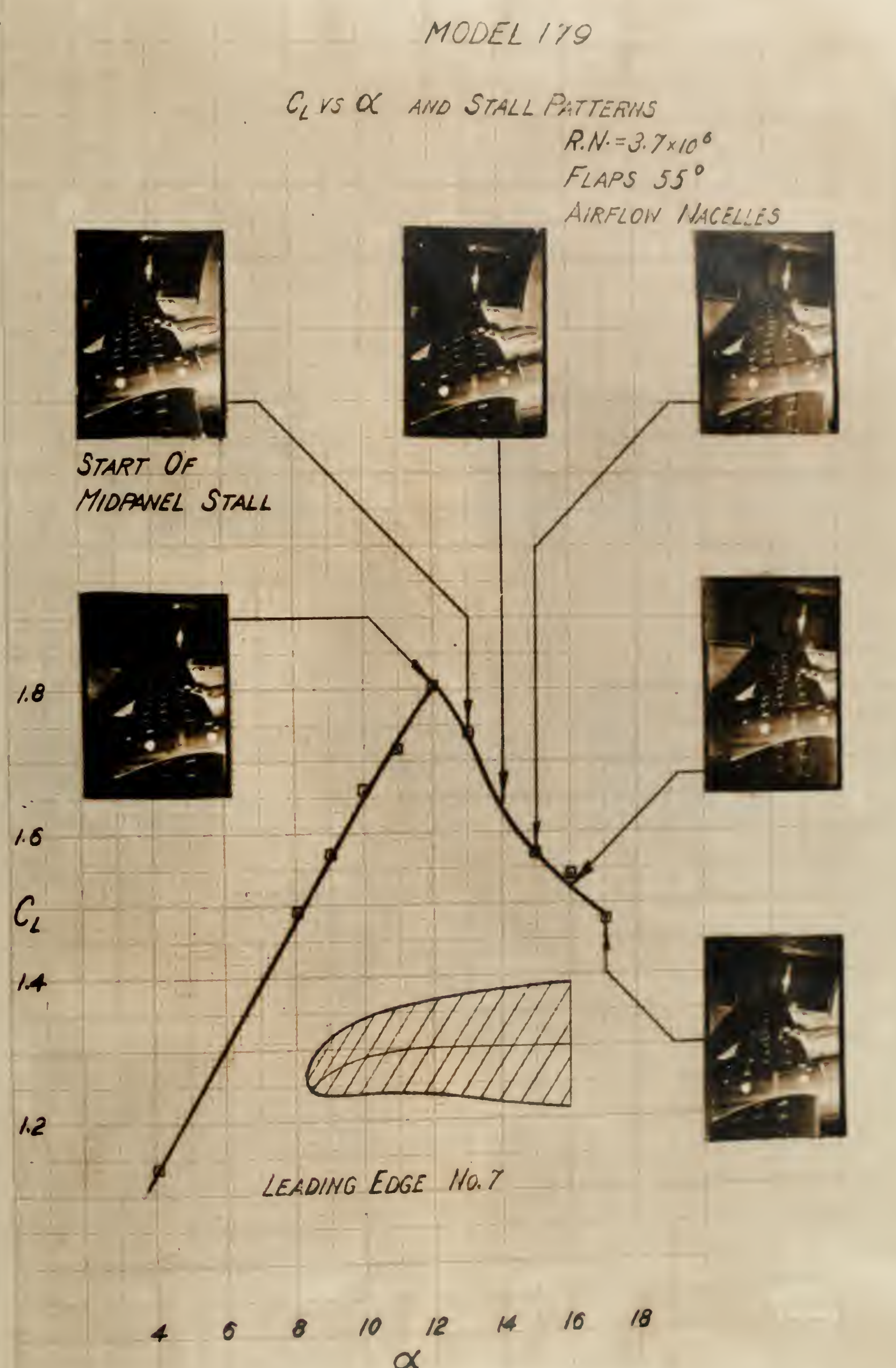
MODEL 179
C_L VS α AND STALL PATTERNS
R.N. = 3.7×10^6
FLAPS 55°
AIRFLOW NACELLES
START OF STALL NEAR TIP
1.8
1.6
C_L
1.4
1.2
LEADING EDGE NO. 5
4
6
8
10
12
14
16
α

MODEL 179

C_L vs α AND STALL PATTERNS

$R_N = 3.7 \times 10^6$
FLAPS 55°
AIRFLOW NACELLES

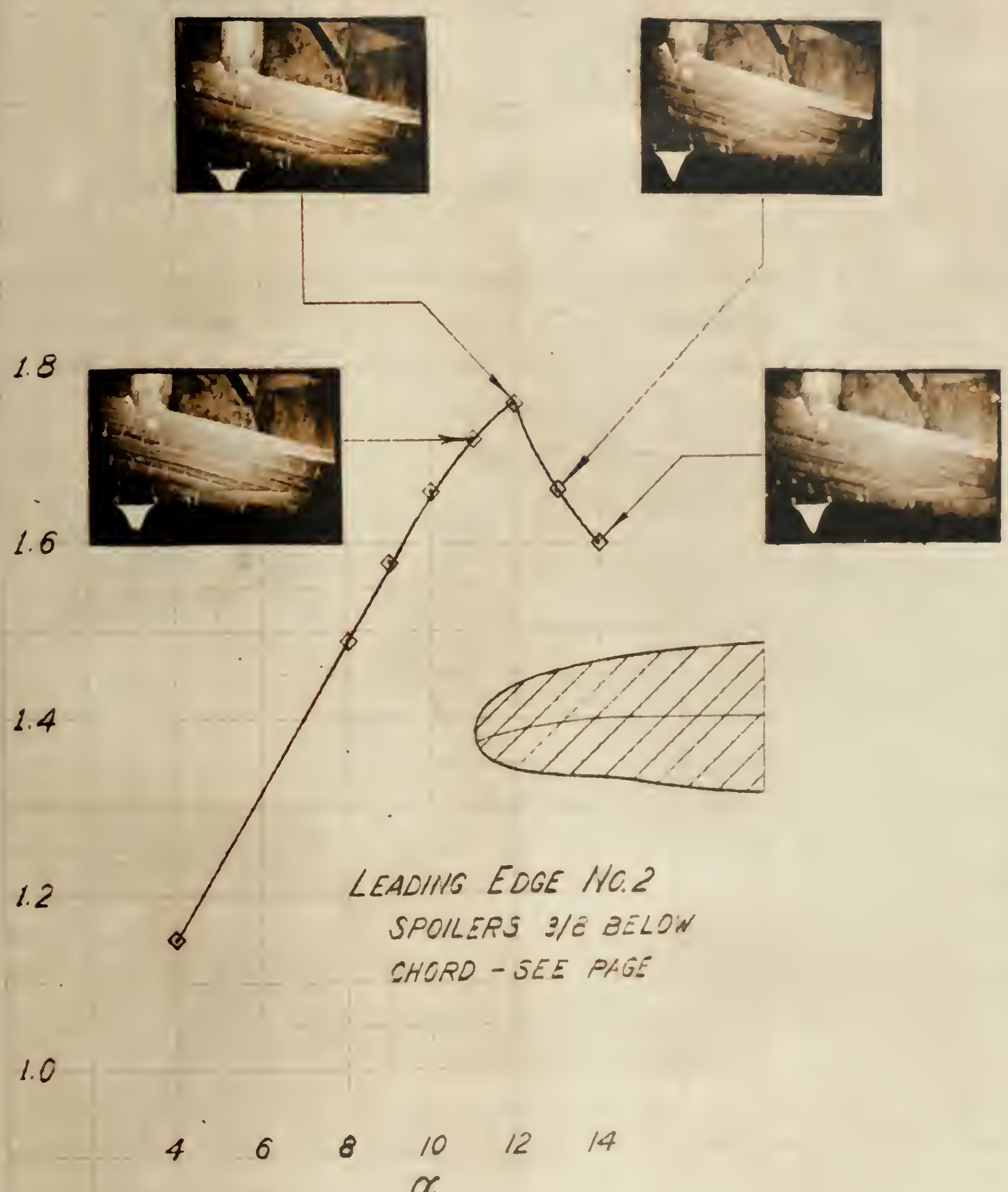



MODEL 179
C_L VS α AND STALL PATTERNS
R.N. = 3.7×10^6
FLAPS 55°
AIRFLOW NACELLES
START OF
MIDPANEL STALL
1.8
1.6
C_L
1.4
1.2
LEADING EDGE No. 7
4
6
8
10
12
14
16
18
α

MODEL 179

C_L VS α AND STALL PATTERNS

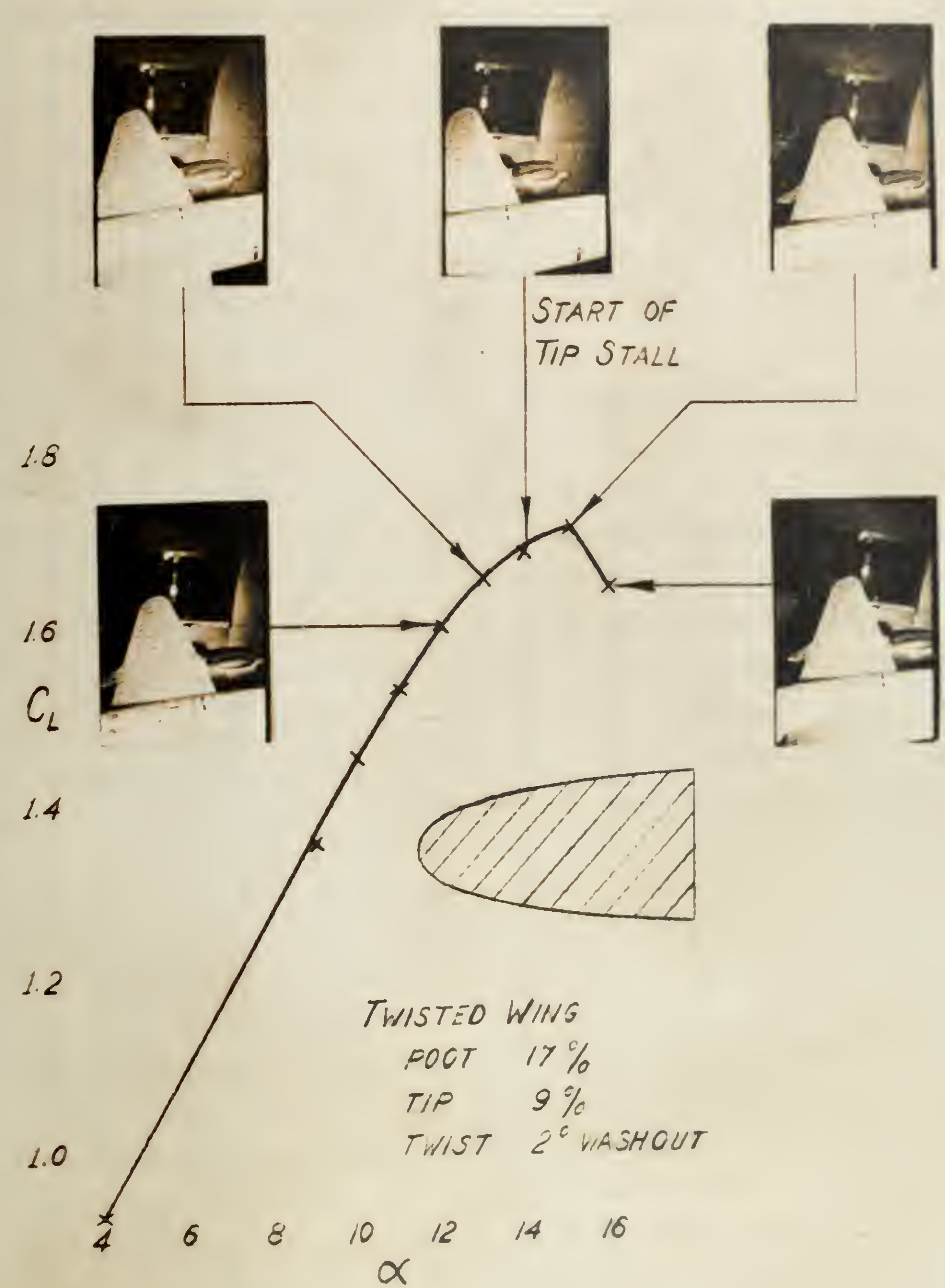
R.N. 3.7×10^6
FLAPS 55
AIRFLOW NACELLES

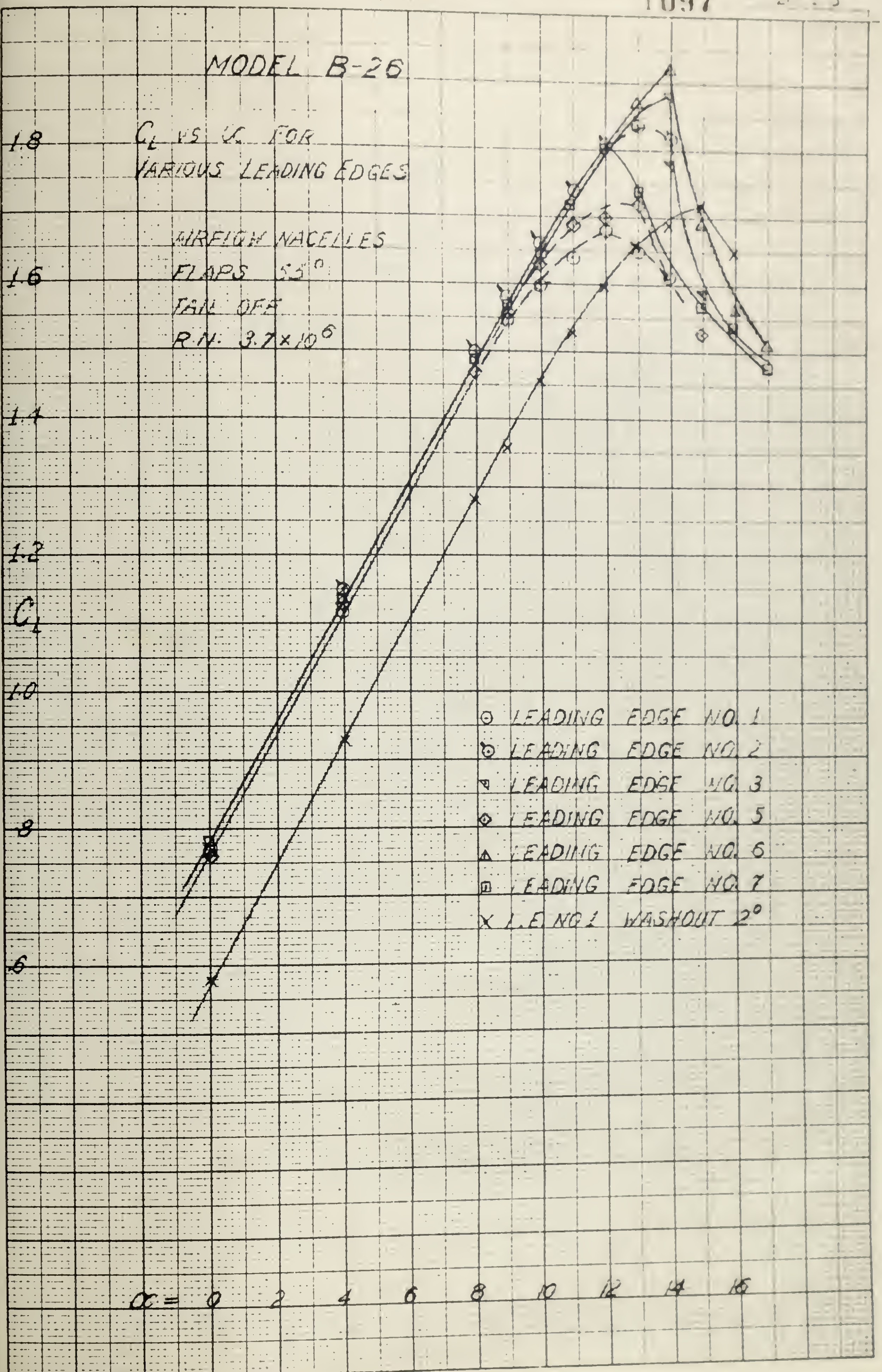


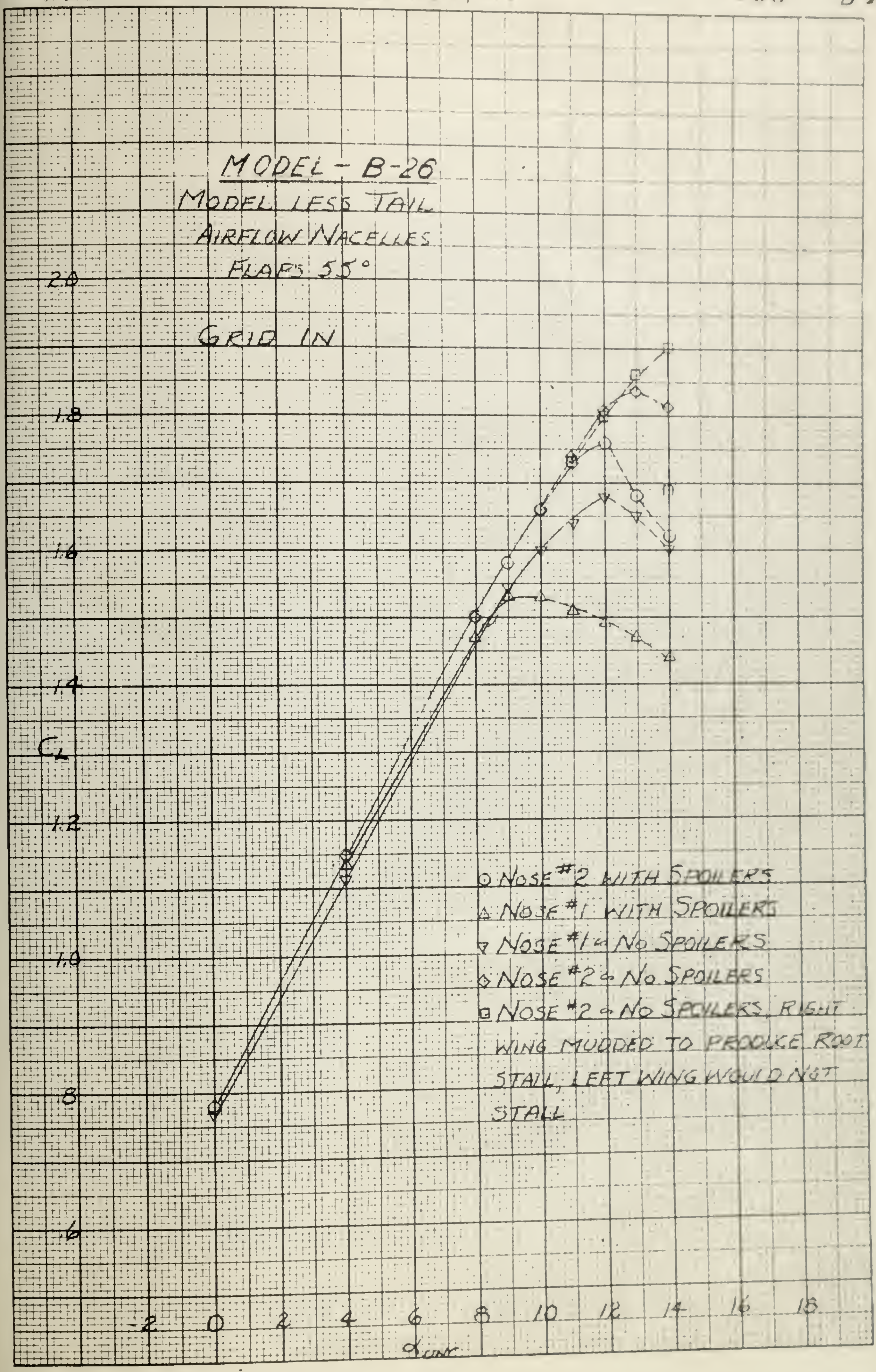
MODEL 179

C_L VS α AND STALL PATTERNS

R.N. = 3.7×10^6
FLAPS 55°
AIRFLOW NACELLES




MODEL B-26
C_L VS α FOR
VARIOUS LEADING EDGES
AIRFLOW NACELLES
FLAPS 55°
TAIL OFF
R.N. 3.7×10^6
LEADING EDGE NO. 1
LEADING EDGE NO. 2
LEADING EDGE NO. 3
LEADING EDGE NO. 5
LEADING EDGE NO. 6
LEADING EDGE NO. 7
L.E. NO 1 WASHOUT 2°
1.8
1.6
1.4
1.2
C_L
1.0
.8
.6
α = 0 2 4 6 8 10 12 14 16

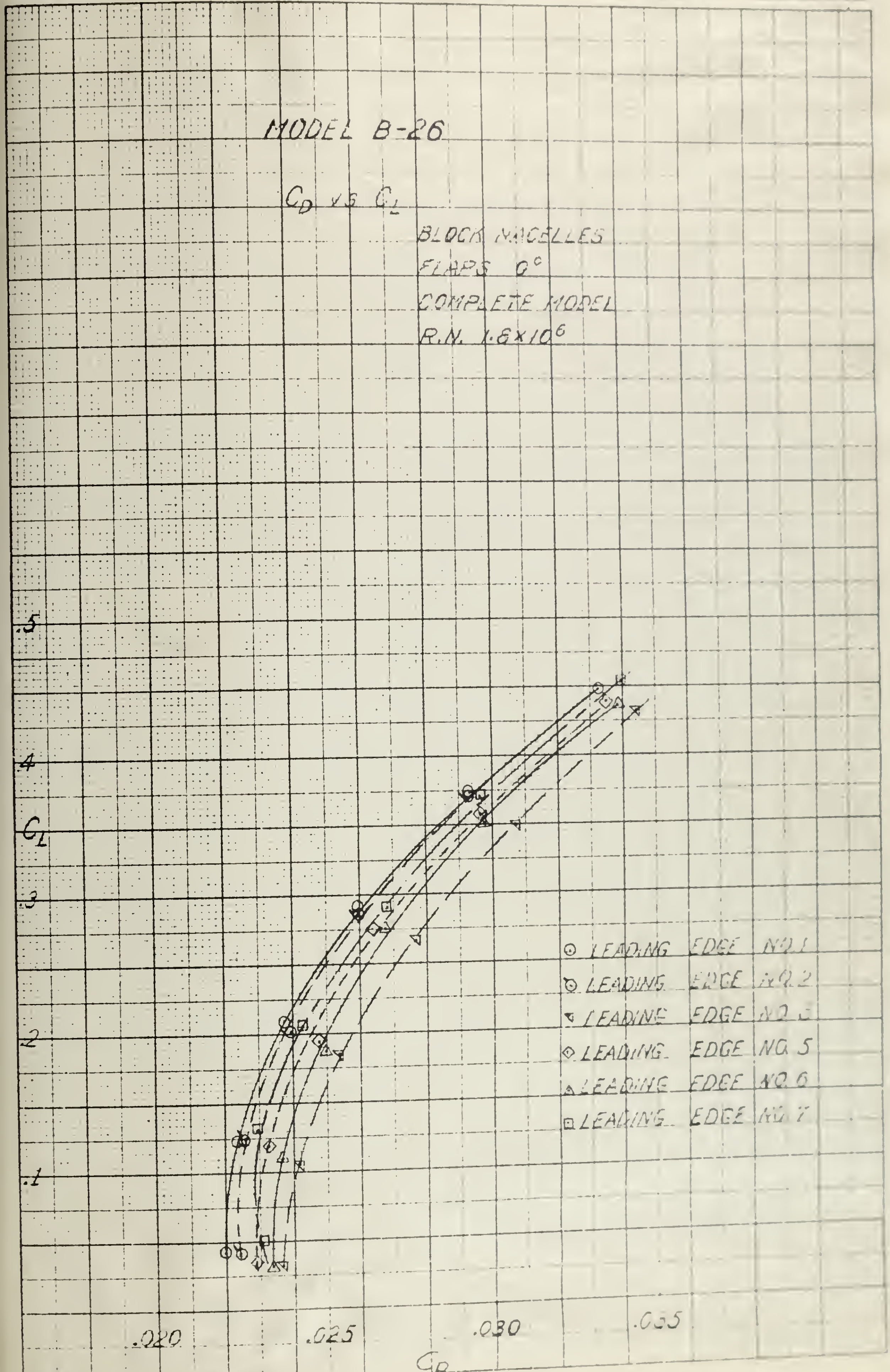

MODEL - B-26
MODEL LESS TAIL
AIRFLOW NACELLES
FLAPS 55°
GRID IN
C_L
2.0
1.8
1.6
1.4
1.2
1.0
.8
.6
-2
0
2
4
6
8
10
12
14
16
18
α_{unc}
NOSE #2 WITH SPOILERS
NOSE #1 WITH SPOILERS
NOSE #1 - NO SPOILERS
NOSE #2 - NO SPOILERS
NOSE #2 - NO SPOILERS, RIGHT WING MUDDED TO PRODUCE ROOT STALL, LEFT WING WOULD NOT STALL

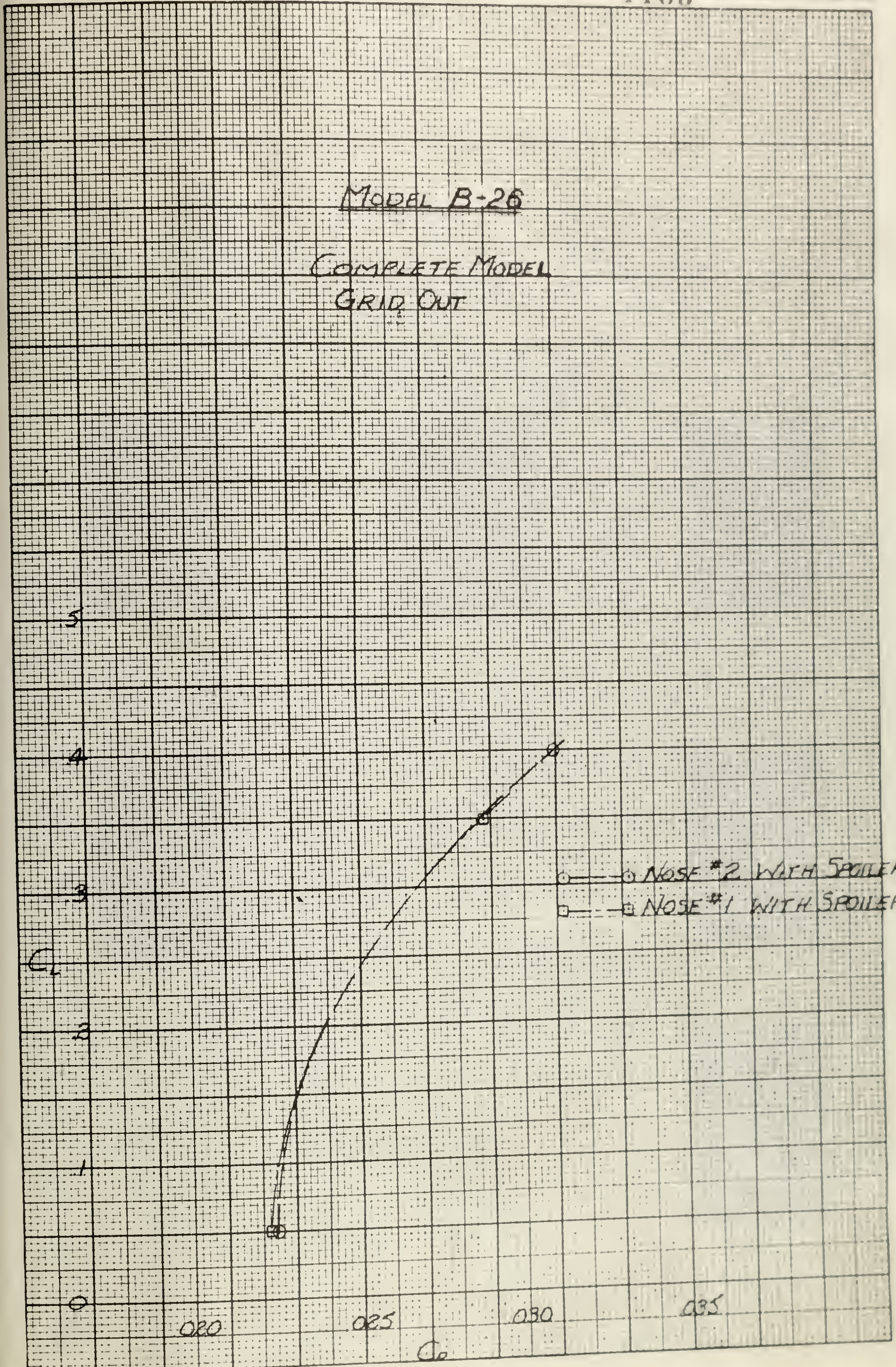
Checked By Cooper
7-18-40

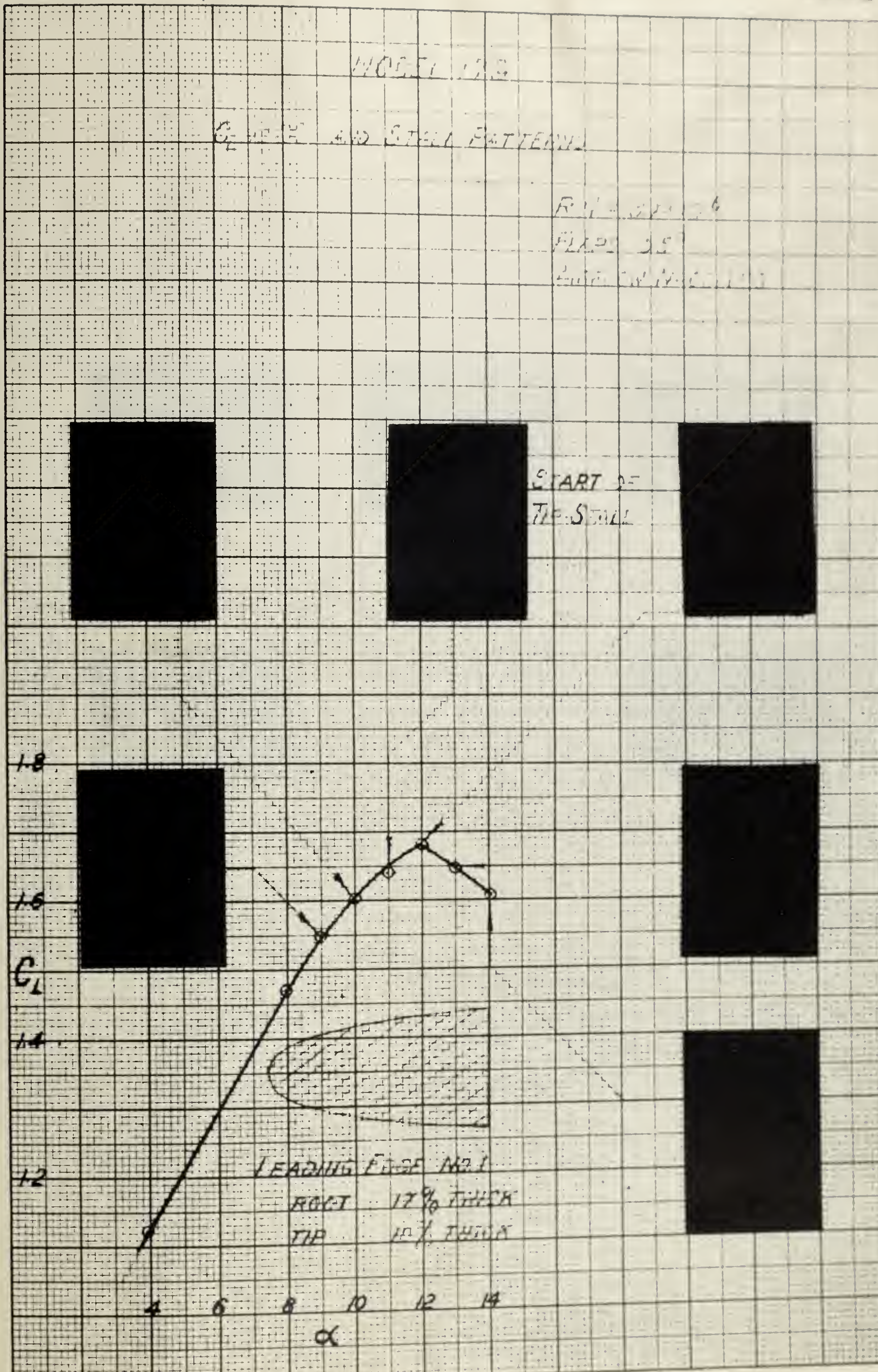
THE GLENN L. MARTIN COMPANY
BALTIMORE, MD.

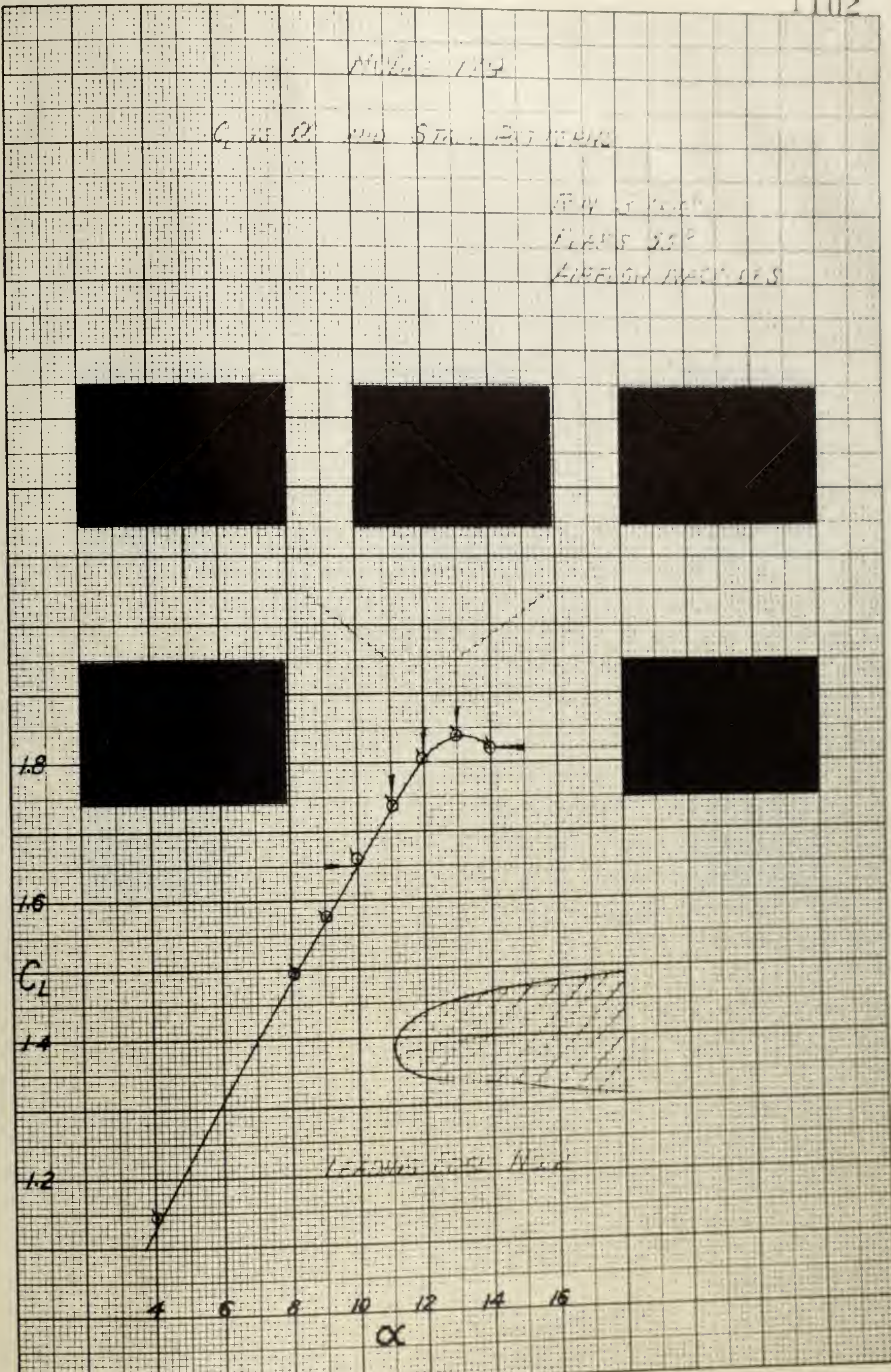
1099

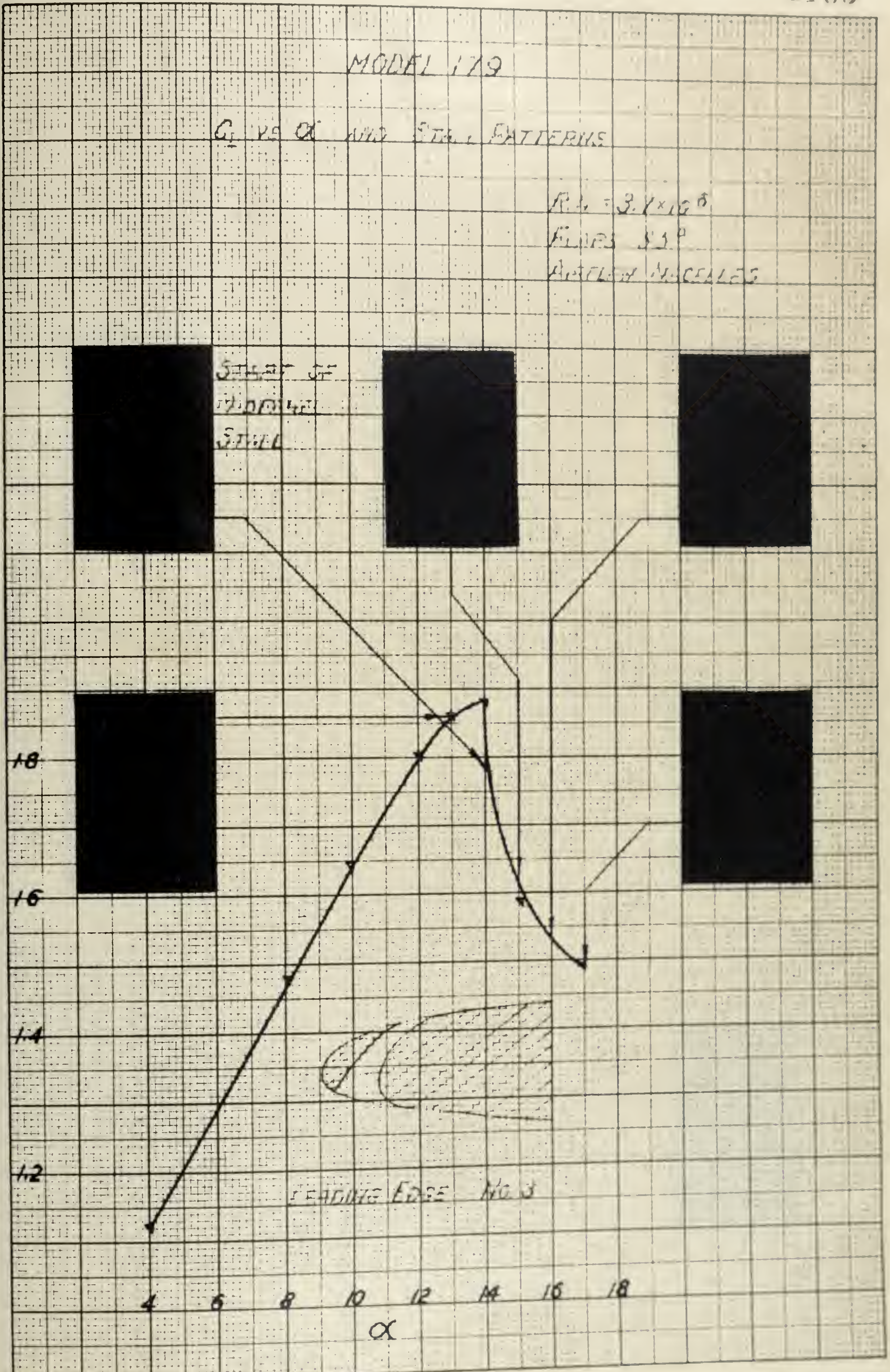
1326
B-26

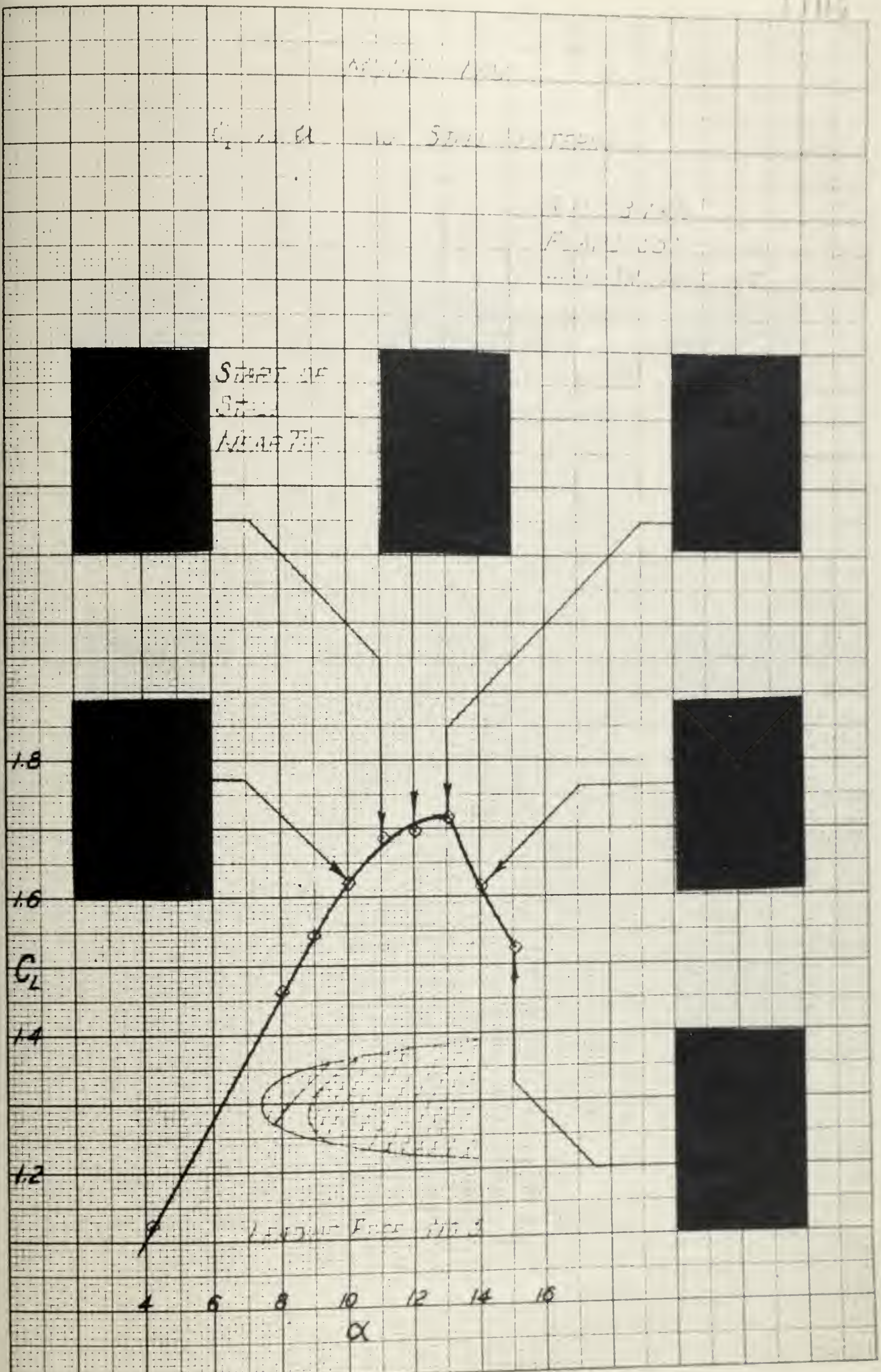


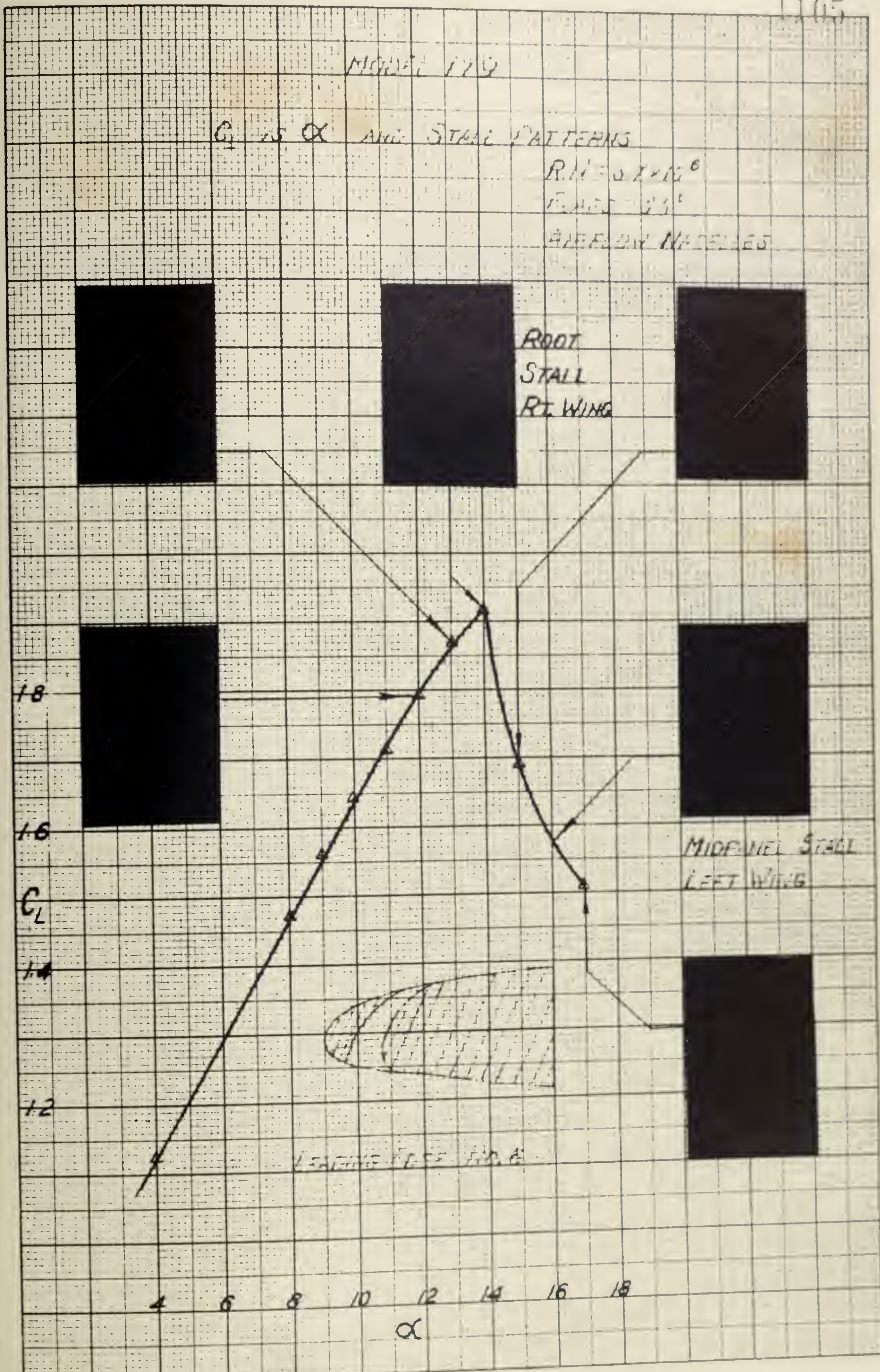

MODEL B-26
COMPLETE MODEL
GRID OUT
NOSE #2 WITH SPOILERS
NOSE #1 WITH SPOILERS
C_L
C_D
0
.1
.2
.3
.4
.5
.020
.025
.030
.035

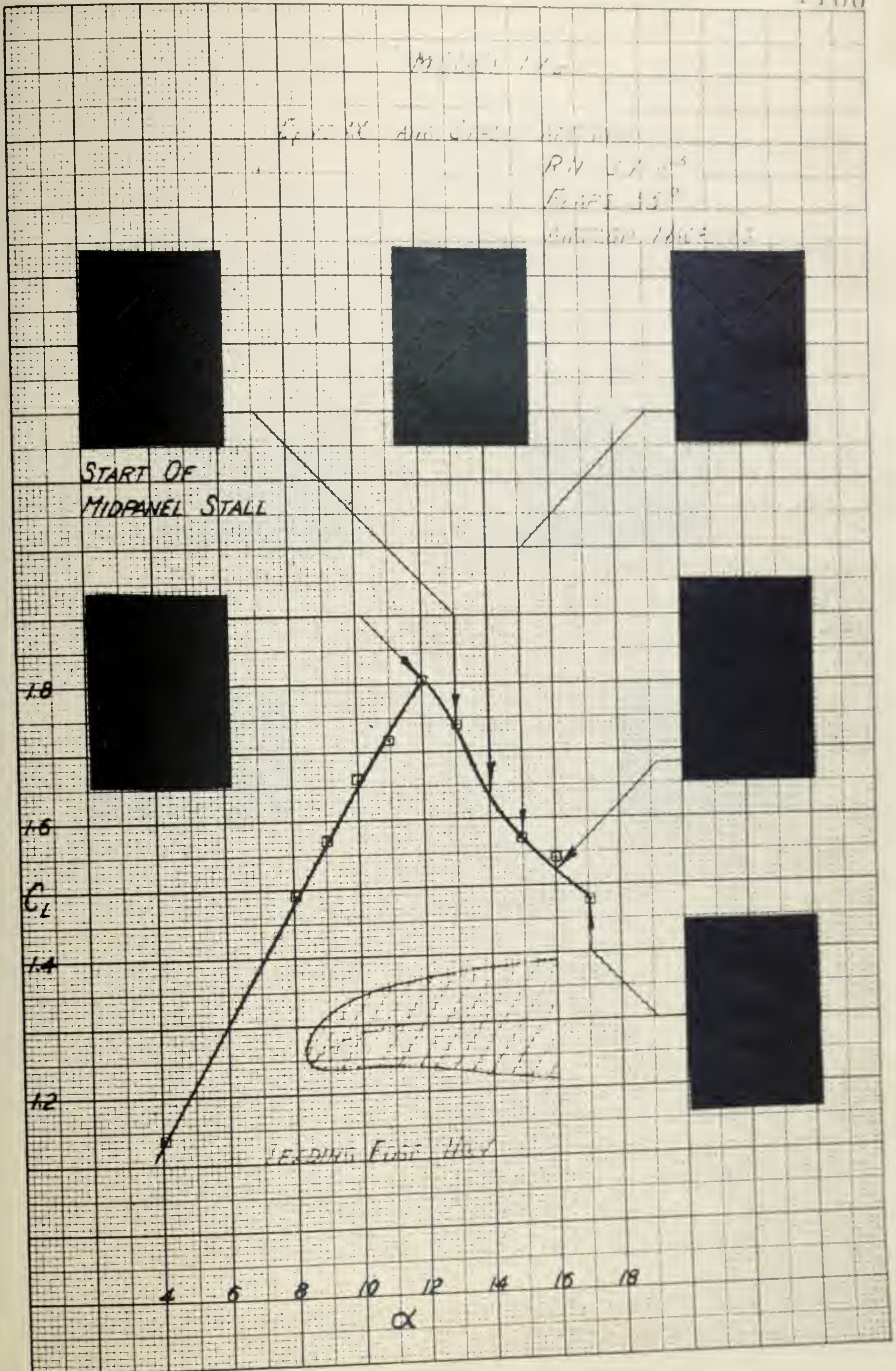

START OF
TIP STALL
1.8
1.6
C_L
1.4
1.2
LEADING EDGE No.1
ROOT 17% THICK
TIP 10% THICK
4
6
8
10
12
14
α


1.8
1.6
C_L
1.4
1.2
4
6
8
10
12
14
16
α


MODEL 179
1.8
1.6
1.4
1.2
4
6
8
10
12
14
16
18
α


1.8
1.6
C_L
1.4
1.2
4
6
8
10
12
14
16
α

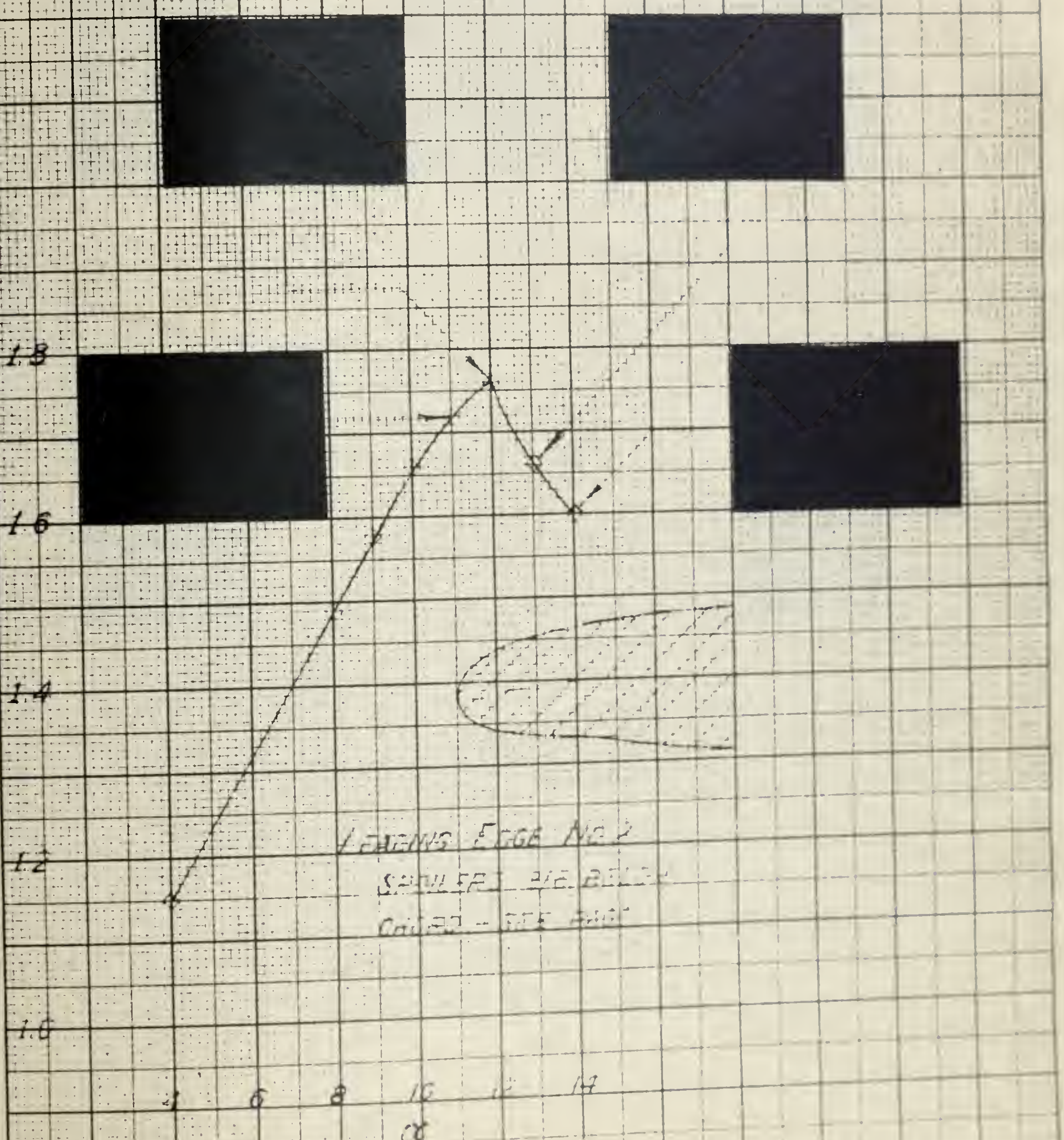

STALL PATTERNS
ROOT STALL RT WING
MIDPANEL STALL
LEFT WING
1.8
1.6
C_L
1.4
1.2
4
6
8
10
12
14
16
18
α

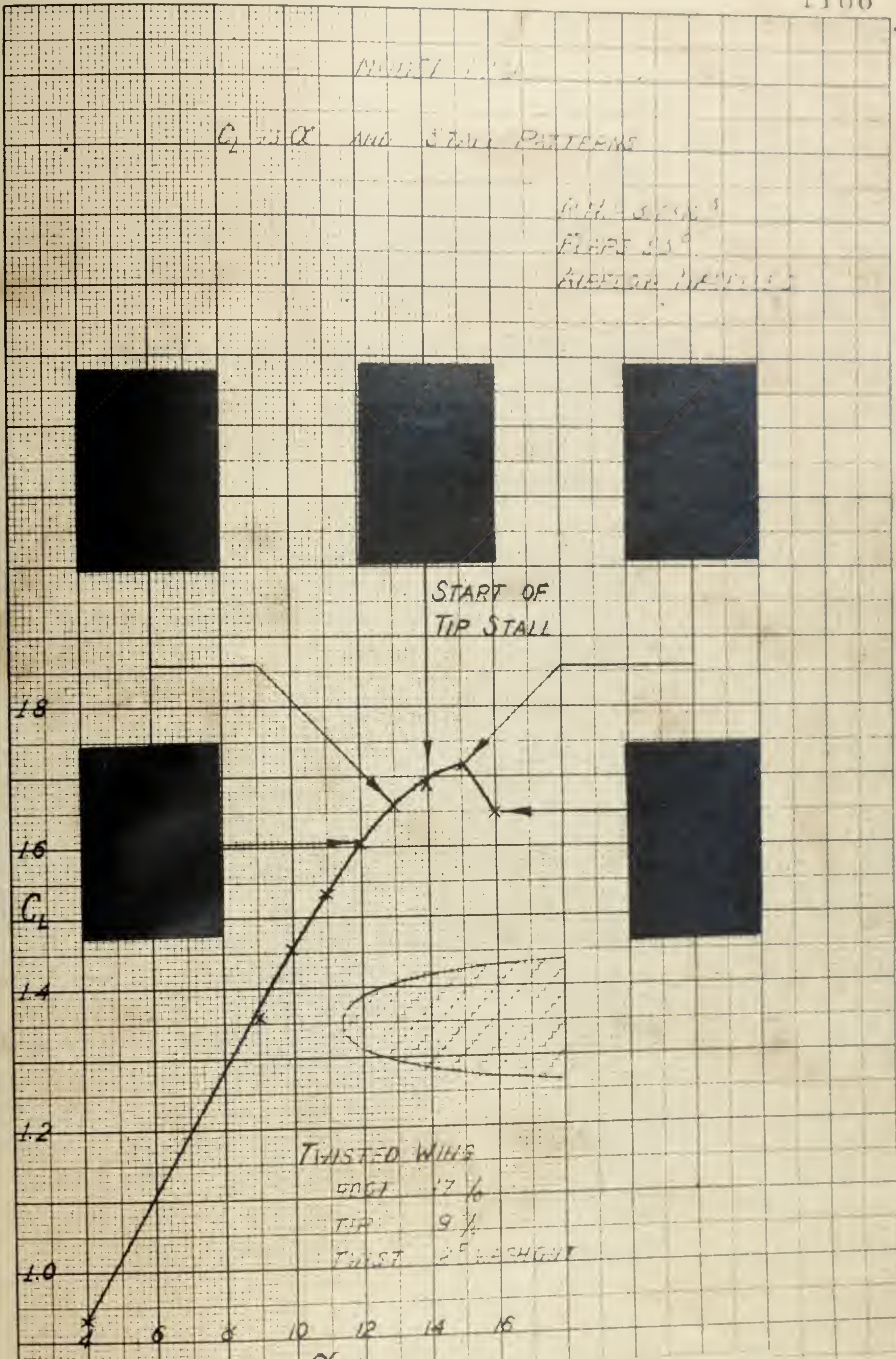

START OF
MIDPANEL STALL
1.8
1.6
1.4
1.2
C_L
4
6
8
10
12
14
16
18
α

MODEL IV9

C_L vs α and Stall Patterns

R.N. 3.7×10^6
Flaps 55
Airflow Nacelles





Admitted [illegible] 24, [illegible].

In the United States Court of Appeals
for the Ninth Circuit

No. 12885

CONSOLIDATED VULTEE AIRCRAFT CORPORATION, a Delaware Corporation, and AMERICAN AIR LINES, INC., a Delaware Corporation,

Appellants,

vs.

MAURICE A. GARBELL, INC., a California Corporation, and GARBELL RESEARCH FOUNDATION, a California Corporation,

Appellees.

STIPULATION AND ORDER

It Is Hereby Stipulated by and between appellants and appellees under the provisions of Rule 76 (h) that the record on appeal may be supplemented to include the following material omitted from the record on appeal:

(a) The affidavit of Theodore Roche, Jr., executed January 31, 1951, filed in opposition to Defendants' Motion for New Trial.

(b) The affidavit of Maurice A. Garbell, executed January 30, 1951, filed in opposition to Defendants' Motion for New Trial.

(c) This stipulation.

And that this stipulation constitute a designation of the supplemental record to be printed as a supplement to the record heretofore filed in this cause and that the attached constitute true copies of the affidavits of Theodore Roche, Jr., and Maurice A.

Garbell hereinabove identified and that the supplemental record so designated by this stipulation may be printed and will constitute a supplement to the record on appeal.

This stipulation is entered into at the request of appellees, and appellants consent thereto only upon the condition that their time for filing their opening brief on appeal be continued and reset to commence upon the clerk's mailing to appellants copies of the printed supplement to the printed record referred to in this stipulation, such time to expire not earlier than September 16, 1951.

It Is Further Stipulated that the cost of printing the supplement referred to herein shall be borne by appellees.

Dated August 14, 1951.

HARRIS, KIECH, FOSTER & HARRIS,
/s/ FORD HARRIS, JR.,
Attorneys for Appellants.

LYON & LYON,
/s/ LEWIS E. LYON,
Attorneys for Appellees.

So Ordered:

/s/ WILLIAM DENMAN,

Judge of the United States Court of Appeals for the Ninth Circuit.

/s/ CLIFTON MATHEWS,

/s/ H. T. BONE,

Judges U. S. Court of Appeals for the Ninth Circuit.

District Court of the United States, Southern District of California, Central Division

Civil Action No. 10930-Y

MAURICE A. GARBELL, INC., a California Corporation, and GARBELL RESEARCH FOUNDATION, a California Corporation,

Plaintiffs,

vs.

CONSOLIDATED VULTEE AIRCRAFT CORPORATION, a Delaware Corporation, and AMERICAN AIR LINES, INC., a Delaware Corporation,

Defendants.

AFFIDAVIT OF THEODORE ROCHE, JR.

State of California,
County of Los Angeles—ss.

Theodore Roche, Jr., being first duly sworn, deposes and says:

That at all of the times herein mentioned affiant was, and is now, one of the attorneys of record for the plaintiffs in the above-entitled action, and as such has read and is familiar with defendants' Motion for a New Trial, together with supporting affidavits hereinbefore filed herein.

Addressing himself to the grounds of said Motion for New Trial of (1) surprise and (3) newly discovered evidence, affiant states the following:

Trial of this action commenced at 10:00 a.m., Tuesday, November 21, 1950. Prior thereto and on the 3rd day of August, 1950, by stipulation, defendants took the deposition of Maurice A. Garbell, the inventor of the patent involved herein. Said witness was questioned by Mr. Fred Gerlach, one of defendants' attorneys of record, who at all times during the taking of said deposition was assisted by Mr. Glendon T. Gerlach, the patent director of Consolidated Vultee Aircraft Corporation.

No restrictions or limitations of any kind were placed upon the examination of said Maurice A. Garbell, and said defendants, through their counsel, were afforded full opportunity to, and had they so desired, could have questioned the said Maurice A. Garbell fully, completely and in detail concerning all of the matters, and each of them, ultimately testified to by him during the trial of said action, and by such questioning could have ascertained the name and whereabouts of each person referred to by Mr. Garbell in said testimony, including those individuals named in defendants' Motion for New Trial, to wit: Harry B. Chin, Theodore P. Hall and Donald A. Hall.

This action was commenced in January, 1950. Long prior thereto defendants fully knew that the said Theodore P. Hall and Donald A. Hall were employed by defendant, Consolidated Vultee Aircraft Corporation during the entire period of employment of Maurice A. Garbell by said last named defendant, and that the said Theodore P. Hall and Donald A. Hall were possessed of knowledge which had direct bearing upon the activities of the said

Maurice A. Garbell during the period of his employment by defendant, Consolidated Vultee Aircraft Corporation, with relation to the subject matter of the invention referred to herein.

In the year 1948 affiant was engaged in investigating the truth or falsity of the facts as related to him by Maurice A. Garbell in order to determine whether or not to accept employment in a proposed action against defendants herein based upon the alleged infringement of the patent in suit. In the course of such investigation affiant had several conferences with Mr. Glendon T. Gerlach, who then was and is still Patent Director for Defendant, Consolidated Vultee Aircraft Corporation. Among other things affiant informed Mr. Gerlach that Mr. Garbell claimed that he had suggested the use of his patented wing to defendant, Consolidated Vultee Aircraft Corporation, at every opportunity during a period commencing within a few weeks after the start of his employment by said defendant and lasting during the entire term of said employment.

On or about the 21st day of July, 1948, at the prior suggestion of the said Mr. Glendon T. Gerlach, affiant visited the plant of defendant, Consolidated Vultee Aircraft Corporation, at San Diego, California, the said Mr. Gerlach and affiant together then interviewed and questioned the following persons: Theodore P. Hall, Donald A. Hall, Ralph Bayless and Kenneth E. Ward, all of whom were then and there working at the said plant of defendant, Consolidated Vultee Aircraft Corporation. Each individual was interrogated by affiant and by Mr. Gerlach as to his knowledge of the patented Garbell wing and the suggestions of its use as made by Mr. Garbell during the term of his employment by defendant, Consolidated Vultee Aircraft Corporation.

Upon the conclusion of said interviews, the said Glendon T. Gerlach made a statement to affiant in substantially these words:

"At the outset I was sure Garbell had never mentioned his wing, but after hearing the men today I am convinced Garbell tried to push the use of his wing at every opportunity."

In the early part of August, 1948, the said Mr. Gerlach and affiant had a further interview with Donald A. Hall at the plant of said defendant, Consolidated Vultee Aircraft Corporation, relating to the same subject matter. During one of the conferences held between affiant and the said Glendon T. Gerlach there was placed in affiant's hand by Mr. Gerlach a copy of an analysis of the Garbell patent, which analysis was signed by the said Donald A. Hall and which has been introduced in evidence in this case by plaintiffs as their Exhibit 21.

The Ralph L. Bayless and Kenneth E. Ward above referred to testified on behalf of defendants at the trial of this action. The said Glendon T. Gerlach, Patent Director of defendant, Consolidated Vultee Aircraft Corporation, assisted defendants' counsel in the preparation and trial of this action and he had actual knowledge of the connection between Mr. Garbell, the inventor, Theodore P. Hall, Donald A. Hall and defendant, Consolidated Vultee Aircraft Corporation, as hereinabove set forth. Defendants did not call the said Theodore P. Hall and Donald A. Hall as witnesses.

On the 3rd day of July, 1948, while investigating the facts of this case as hereinabove set forth, for the first time affiant interviewed Mr. Harry B. Chin, and upon that occasion took a statement from him. It had been explained to Mr. Chin that a statement from him was desired upon the ground that Mr. Garbell was dealing with a potential licensee of his patented wing and that we desired to ascertain if there was proof of invention prior to the employment of Mr. Garbell by the potential licensee. Said statement was taken by affiant at his office, not in the presence of Mr. Garbell, was voluntarily given by Mr. Chin; said statement was taken down in shorthand by the secretary of affiant, thereafter transcribed in the office of affiant, such transcription being as follows:

"Mr. Roche: What is your full name?

"Mr. Chin: Harry Bradford Chin.

"Q. And the address?

"A. My present address is 715 Commercial Street—that is where I pick up all my mail—my family live there; although I have an apartment of my own at 1060 Powell Street.

"Q. At the present time you are employed?

"A. By United Airlines.

"Q. In the San Francisco office?

"A. Yes, at Mills Field.

"Q. In what capacity?

"A. Aerodynamic performance engineer.

"Q. You know Dr. Garbell? A. Yes.

"Q. Do you recall when you first met him?

"A. I first met him when I was working in the Boeing School already, and Dr. Garbell came right after a Mr. Thorpe left, which I guess was in November, or thereabouts—October or November—of 1939, or thereabouts, I believe.

"Q. The Boeing School that you refer to is located here?

"A. Yes, at Oakland Municipal Airport.

"Q. Tell me, was that school established by Boeing Aircraft Corporation, or was it established by the Government?

"A. It was established by—through donations of W. E. Boeing, way back in 1929, before the consolidation of airlines, which was later called United Airlines—part of Boeing Transport and Boeing Air Company—before the mail cancellation in 1934.

"Q. It was established as sort of a foundation, by Boeing personally, from his own funds, I gather?

"A. Yes.

"Q. And the purpose of the school was what? They instructed and——

"A. Yes. You might say it is a trade school, and it is a source from which Boeing Air Transport and National Air Transport and quite a few of the airlines draw their personnel—their mechanical personnel.

"Q. You went there first when?

"A. I went there first as a student in 1934.

"Q. To become an engineer?

"A. At that time there was no engineering course. I took what they call a Master Mechanic course and Design Subjects, and so forth. It is a regular mechanics school as well as a flying school.

"Q. They taught flying also? A. Yes.

"Q. That was in 1934? A. Yes.

"Q. At that time were the air lines drawing on that school for their personnel?

"A. Yes, considerably, because there is a placement bureau opened by Boeing School, which helped the graduate to obtain jobs in the industry.

"Q. Did they charge a student going there? Did he have to pay for his tuition? A. Yes.

"Q. It was a regular trade school, in the accepted sense? It wasn't maintained by the air lines? They didn't pay——

"A. No, it was self-supporting.

"Q. You started there in 1934, and as a student you were at the school how long?

"A. One year. Not quite one year. In fact, the course was a nine-month course and I graduated and then I took a couple of months of postgraduate work, so making it, all in all, eleven months. Then I became an instructor in aeronautics at the same school.

"Q. You became an instructor the latter part of 1934, or 1935?

"A. No. The latter part of 1935.

"Q. And you were instructing in what capacity?

"A. At first—the first few months I was a reader in the Aerodynamics and Strength of Material Department, as well as assistant instructor in Drafting and Designing.

"Q. That latter subject—was that the drafting and designing—was that airplanes or planes as a whole, or wing design or body design?

"A. Generally complete ships.

"Q. Did you continue in that particular field, or did you progress into other subjects, and between 1935 and 1939—there is a four-year period—you remained at the school?

"A. Yes. During those four years, while I taught a variety of subjects, including mathematics and aerodynamics and mechanical design and illustrative and descriptive geometry.

"Q. Along the latter part of 1939 you say there was a Mr. Thorpe. Was he an instructor?

"A. Yes.

"Q. And he left there? A. Yes.

"Q. And Dr. Garbell came to take his place?

"A. Not exactly to take his place; you might say as far as the lecture material was concerned. While the instructors do teach the various material, and when Mr. Thorpe left there was an opening, obviously, and I believe Mr. Garbell was hired on that open requisition.

"Q. There was an opening and he was employed, as far as you know, to fill it?

"A. Yes. Because I took over most of Thorpe's subjects after Thorpe left, which was mainly design.

"Q. Plane design? A. Yes.

"Q. I suppose it is true, Mr. Chin, that during those four years, in connection with the field of plane design, that you had given a lot of attention and a considerable part of your work dealt with wing design, and the structure of wings, and the air forces? That is true, isn't it?

"A. That is right.

"Q. Did you teach or lecture students on those subjects?

"A. Yes. Simultaneously during those four years Mr. Thorpe and myself designed two airplanes for Boeing School of Design, and both those airplanes were built by the school and flown by the school.

"Q. What type, single motor?

"A. Single-motor, two-seater trainers.

"Q. Did you embody any new principles of design in those planes, either in the wing, or in any fashion, from what had preceded the trainers? There was some change, wasn't there?

"A. There are changes going on at all times, due to past knowledge. I would say the airplanes that we designed were strictly conventional types, because we designed it as a trainer; so, therefore, any characteristics of the airplane should be, of necessity, conventional, and those characteristics are known; so that the airplane, when done, would be an honest, conventional, othodox airplane?

"Q. There wouldn't be any radical change in it, then. Is that correct? A. Yes.

"Q. But in the field of aerodynamics and the designing of planes, and wings in particular—during those four years you gave great study to different types of wing construction? A. Yes.

"Q. And principally dealing with the effect of air flow over the wing surface, isn't that correct?

"A. Yes.

"Q. Now at the time there already had been some well-known patents issued on or covering wing design? A. Yes, there were.

"Q. For many different types of wings; but there were some major types which were in general use. Isn't that correct? A. Yes.

"Q. And you were familiar with them?

"A. Yes.

"Q. Now, tell me, Mr. Chin, it is true, isn't it, that the goal toward which a plane designer goes is to design a wing which has very good stalling characteristics? A. Yes.

"Q. In other words, everybody designing a plane, or a wing for a plane, for general use attempts to eliminate, if they can, stalling characteristics. Is that right?

"A. No. I do not believe that is a correct statement, because I do not believe you can entirely eliminate stalling. You might say we try to eliminate any violent characteristics accompanying a stall, and, if possible, have sufficient warning before a stall.

"Q. In other words—let me put it this way: the result which you would seek to achieve in designing the wing would be that in performance violent results would tend to be eliminated from the stalling characteristics, first—— A. Yes.

"Q. And, secondly, or as a part of it, the design in operation would cause the wing to give a warning that a stall was approaching. Is that correct?

"A. That's right. You might put it that way, more specifically: an airplane that has honest stalling characteristics should be designed such that a stall is unaccompanied by any rolling motion of the airplane, and that can be done—whatever the means is a different story—by moving the point at which the complete wing first stalls—by moving this point inboard, closer to the fuselage. If the initial stalling point is out toward the wing tip, obviously any stall would be accompanied by a rolling motion of the airplane, and if the stall point is inboard or closer to the fuselage of the airplane, then when the airplane does stall, it will stall and fall straight ahead, unaccompanied by an violent rolling motion. In other words, it will just pitch until its usefulness is again obtained, by pitching of the nose downwards.

"Q. In other words, the nose would pitch downwards, so that the plane would tend to drop, and thereby gain speed, so that the stall of the wing would be again overcome. A. That's right.

"Q. In the case where the stall is accompanied by a very violent rolling motion and a plane does get into a stall, does a spin result in the plane?

"A. Generally, yes.

"Q. And then that is almost impossible to pull out of, is that right, in these larger planes?

"A. Not exactly; if the airplane is what we call dynamically stable, the airplane will come out of a spin, with the controls neutral, by itself within 1½ rolls. You might say if the airplane has gotten into a spin and the controls are neutralized, and the hands and feet are off the controls, the airplane should be able to pull out of a spin within 1½ rolling motions of the airplane—and by itself; if the airplane were dynamically stable. Of course, you do have the catastrophic type, that gets worse and worse as it spins, but if the airplane were designed correctly it would come out of it.

"Q. Had you known Dr. Garbell, or known or heard of him, prior to his coming to the Boeing School? A. No, I had not.

"Q. So that the first time you ever heard of him or met him was after he became employed as an instructor at the school? A. Yes.

"Q. This was in 1939? A. Yes.

"Q. At that time, with the general world conditions being what they were, had the government stepped into the picture in any way in connection with that school? A. No.

"Q. However, due to certain security rules and regulations which were in existence, a person who was not a citizen of the United States could not work or have any connection with any of the airplane manufacturers and buiders at that time. Is that true? Were you aware of that?

"A. I believe that has been the practice of all the major companies, to hire only citizens or persons who have taken out first papers.

"Q. Let's say this: Dr. Garbell was at the school from October or November of 1939, according to your recollection, until when? About?

"A. He left to join Pan-American Air Ferry Group, let us say—I would say somewhere in 1941.

"Q. To the best of your recollection, in round numbers, he was at the school approximately two years, we will say. That is correct? A. Yes.

"Q. During that two-year period, did you become acquainted with the doctor? A. Yes.

"Q. Were you working together in connection with any projects of the school?

"A. Not any particular project, no; but as far as teaching courses, yes. In other words, for instance, when Dr. Garbell left—he left in the middle of the semester, as it were—I took over a couple of his courses.

"Q. What courses did you take over?

"A. I took over the differential calculus course from him and also the strength of materials course from him.

"Q. Did Dr. Garbell teach or lecture in connection with a course on plane or wing design, do you recall?

"A. I was the chief instructor in design at the time he left, although Dr. Garbell taught some aerodynamic courses—which ones I don't recall.

"Q. Now, tell me, Mr. Chin, during the time that Garbell was at the school there, did you become pretty well acquainted with him? A. Yes.

"Q. And I suppose that in connection with your work you had frequent discussions of problems, is that right? A. Yes.

"Q. And, of course, you both were interested in everything connected with aerodynamics and planes, that's true? A. Yes.

"Q. And during that time I suppose you had many, frequent discussions and conversations concerning problems in a general way—unconnected with your school work, as we may say; in other words, looking at the aerodynamic field in its broad plane. Is that right? A. Yes.

"Q. Now, tell me something, Mr. Chin: At any time while Garbell was at the school during that two-year period, did you ever hear him discuss, or did he ever tell you anything about, a wing design which he had conceived, which had good stalling characteristics and consisted of a three-section wing, wherein the air foils were changed in some fashion, or any fashion, from what might be said to be the standard arrangements?

"A. He mentioned to me a certain principle that could apply to accomplish the same thing that I was trying to work out in order to get a different principle, which is not completely unorthodox, on which certain information were available already from NACA reports.

"Q. The information from the NACA reports which you just referred to was made available when?

"A. The NAC Reports were made available at all times because the school subscribes, and I myself, personally, subscribed to it, and those reports come in periodically.

"Q. The information you refer to dealt with what particular subject—calibration of air foils?

"A. No, it dealt with—it isn't covered by just one report, it is covered by several reports. One is on the effect of lift characteristics as a function of the Reynolds number.

"Q. The Reynolds number relates to air foils, doesn't it?

"A. It relates to air foils in this way: it has to do with scale effect or size of the air foil; the scale effect of an airplane as compared with that of a tested air foil.

"Q. At that time, did these NACA reports have worked out what you might say the family curve of the airfoils? A. Yes.

"Q. From what you have said, I understand that for some time you yourself had been attempting to work out some principle of wing design utilizing the information in these reports?

"A. Yes, that's right.

"Q. In connection with your work on that idea, had you considered changing the scale, or graduating, I may say, the scale of the air foils from one section of the wing to the other?

"A. The size, or length, of the air foil, of necessity, does change, because of the root length, due to the plane—the form of the wing, long at the root and shorter at the tip; but at no time did I try to change the camber of the air foil not related to the same family. If I started, let us say, at 23,000 c's, I retained 23,000 c's right to the tip. The only variation is on the width of the root.

"Q. I follow you. Under your plan, the corresponding points of the different air foils would be connected by straight lines?

"A. That is right. That is exactly what I am trying to do, to get away from complicated structure.

"Q. I suppose that, with this in your mind, it was a natural consequence that you eventually got into a discussion of these principles with Dr. Garbell?

"A. Yes. In fact, I had worked out the data already, showing that if I twist the wing tip, using the same family air foils, 3 degrees, I would have moved the stall inboard, which is a conventional method, using this NACA information which I have just mentioned, because it was a function of the Reynolds number.

"Q. When you mentioned this to Garbell, of course, that sort of opened the door for a discussion of these matters? A. That's right.

"Q. And at that time did he say that he had worked out the principles to be used in a wing design?

"A. Yes. He mentioned in this broad sense, in the way of conversation, that the same thing I was trying to accomplish could be done by a different method which he had worked on before. But I have not seen any detail of the work, although he mentioned that the end result could be accomplished by a different principle.

"Q. Do you recall at any time, in conversations with you, or in any lecture, or anything like that, that Dr. Garbell referred to a three-section wing utilizing these principles?

"A. I do not recall the number of sections, but he mentioned to me that it could be done by switching sections—that means switching the family relation of the air foil; but I do not recall how many sections it required, how many switches it required.

"Q. You do recall he mentioned that this same thing could be done in more than one section on a wing? A. That is right.

"Q. And by switching the family curve of one section as distinguished from another, the two sections, or as many sections as there were, would differ, one from the other? A. Yes.

"Q. Now, the purpose of utilizing that principle, I take it, would be to move the stalling point of the wing inboard from the tip?

"A. That's right.

"Q. Particularly away from the ailerons?

"A. You don't have to move all the way inboard; and the ailerons generally covering the tip point of the wing—moving them inboard so as to permit a certain degree of control over the ailerons even during stalling.

"Q. You do recall the doctor saying that he had already worked that principle out?

"A. I have not seen any detail of the work, but he did say that he had worked on that particular principle. Let us put it that way.

"Q. Did you ever learn that prior to that time he had utilized this principle in connection with the construction of gliders that had flown? Do you recall that?

"A. I don't recall that he had built one—whether he did say that he had built one—but in other conversations he mentioned that he had built gliders before, his being a captain of the Italian Olympic Glider Team, or something like that; but I don't recall definitely whether he had actually used this particular principle in any of the gliders he may have built; but I do recall, in many other conversations, that he had built gliders before; whether he had applied that principle or not, I do not know, because when he mentioned this particular switching of wings to me, I agreed with him at that time, offhand it sounded all right, but my comment at that time was that probably it would give considerable structural difficulty in not having to pass straight lines between corresponding points on the wings, and complications would arise, wing jigs, and things of that sort. It is a mechanical problem, an aerodynamic problem.

"Q. Of course, such a wing, being built in sections, with different family curves in connection with the air foils, would present, I might say, a broken-line appearance of the completed wing, as distinguished from a straight-line appearance, from fuselage to wing tip. Is that correct?

"A. Yes. You see, I taught descriptive geometry and drafting, along with the design course, and anything complicated like that I would immediately see a structural or mechanical problem that would be difficult to overcome, so I did not pursue it any further, with the discussion we had, nor did I even try it myself, because of the mechanical difficulty that I would see.

"Q. Nothing was worked out? A. Yes.

"Q. In these discussions, Mr. Chin, where you and Dr. Garbell were conversing about this particular wing, would you say that those conversations were had in 1940?

"A. Yes. I would say during 1940—about that time.

"Q. Did they occur upon more than one time, or was the subject referred to now and then——

"A. It was never a continuous discussion, you might put it that way. Oh, maybe one or two other discussions after that. But I did recall this one particular time, where I had just completed my study of using the NACA data, at which time we discussed it, you might say, after hours, an hour; maybe a couple of times afterwards, maybe get ten or fifteen minutes of general discussions; but I don't recall that we pursued that discussion much further, because we had other problems to discuss.

"Q. At any time during these discussions, when the subject was mentioned, did Garbell use any figures or refer to any formula in connection with this principle? A. I do not recall.

"Q. You have no such recollection?

"A. I have no recollection on that, although, quite naturally, I talked with a pencil and paper a lot of the time and he talked with a pencil and paper a lot of the time; but I don't recall that he drew out any particular formula or——

"Q. Do you think during these discussions both of you or one of you drew sketches? Would you say that took place or did not take place?

"A. I would say from my own habits that it probably took place, but I do not recall what we drew.

"Q. Now, Mr. Chin, do you believe that you could state to me, in all fairness, from what you heard Dr. Garbell say, that at that time he had conceived and worked out this principle or a principle of wing design which could be applied to more than one section, so that there would be a dissimilarity of family curve of air foils between one section and the other?

"A. I would say, from the impressions that I had at that time, that he had conceived the idea; but I have no knowledge that he had worked it out completely, because we did not pursue it in any detail, only on the surface——

"Q. Now, I am going to put a question to you somewhat in the nature of considering you as an expert here. Suppose I would say to you that those principles had been used in the construction of a wing placed upon a glider and that the glider had successfully flown. Under those circumstances, would you believe that the principle had been worked out?

"A. Yes, you might say that would be the test or proof.

"Q. As to whether the thing had been worked out or not? A. Yes."

Upon diverse occasions prior to the commencement of the trial of this action affiant requested the said Harry B. Chin to testify upon the trial of said action on behalf of plaintiffs as to the subject matter contained in Mr. Chin's statement as hereinabove set forth. At all times Mr. Chin refused to testify.

Further affiant sayeth not.

/s/ THEODORE ROCHE, JR.

Subscribed and Sworn to before me this 31st day of January, 1951.

[Seal] /s/ FRANCES L. RICHMOND,
Notary Public in and for
Said County and State.

Comm. expires Mar. 7, 1954.

[Endorsed]: Filed February 5, 1951, U.S.D.C.

[Title of District Court and Cause.]

AFFIDAVIT OF MAURICE A. GARBELL

State of California,
County of Los Angeles—ss.

Maurice A. Garbell, being first duly sworn, deposes and says:

That he is the Maurice A. Garbell who has previously testified in the above-entitled cause and that if called to testify further would state as follows:

That he has read the reports referred to in the affidavits of Harry C. Matteson and William W. Fox and the conclusions reached by these men as to what the reports of California Institute of Technology, Galcit Report 504C, dated April 11, 1947, and C.V.A.C. Report ZA-240-008 and C.V.A.C. Flight Test Report of Flight No. 7 of the Model 110 airplane of August 19, 1946, show.

Further, affiant states that these reports do not show the Convair 240 airplance as certificated and sold; that alterations of the nacelles, wing fillets, ailerons and flaps were made to the airplanes as certificated and sold in order to permit the wing to stall as described in the patent in suit, and that in the airplane as certificated and sold the stall inception is over a large inboard area and that this stall progresses inboardward toward the root of the wing and that the stall of said airplane is not a root stall such as I defined a root stall in my testimony.

That Exhibit 35 discloses that after the alterations to the nacelles, the wing fillets, the flaps and the ailerons, the airplane stalled as I have described.

Further, affiant states that the Convair 240 as certificated and sold were airplanes that had modifications made to them to correct the stalling characteristics described in Flight Test Reports No. 6 and No. 7 of Exhibit 35.

/s/ MAURICE A. GARBELL.

Subscribed and Sworn to before me this 30th day of January, 1951.

/s/ IRENE J. KNUDSEN,
Notary Public in and for said County and State above written.

[Endorsed]: Filed February 5, 1951, U.S.D.C.

[Endorsed]: Filed August 25, 1951, U.S.C.A.

[Title of Court of Appeals and Cause.]

STIPULATION AND ORDER

It Is Hereby Stipulated by and between appellants and appellees under the provisions of Rule 76(h) that the record on appeal may be supplemented to include the following material omitted from the record on appeal:

(a) The translation marked as "Exhibit 32a" attached to Defendants' Exhibit AAA; and

(b) This stipulation.

And that this stipulation constitute a designation of the supplemental record to be printed as a supplement to the record heretofore filed in this cause and that the attached constitutes a true copy of said translation "Exhibit 32a" of Defendants' Exhibit AAA hereinabove identified and that the supplemental record so designated by this stipulation may be printed and will constitute a supplement to the record on appeal.

This stipulation is entered into at the request of appellants, and it is further stipulated that the cost of printing the supplement referred to herein shall be borne by appellant.

Dated August 24, 1951.

HARRIS, KIECH, FOSTER & HARRIS,

By /s/ FORD HARRIS, JR.,
Attorneys for Appellants.

LYON & LYON,

By /s/ LEWIS E. LYON,
Attorneys for Appellees.

So Ordered:

/s/ WILLIAM DENMAN,
Chief Judge,

/s/ CLIFTON MATHEWS,
Circuit Judge.

EXHIBIT No. 32a

Translation of Page 419, No. 16

"Flugsport"—1937

Performance Glider FS 16 "Wippsterz"

This plane was designed and built by the "Study-Group for Technology of Airplanes" at the "University for Technology" in Stuttgart. It made its first public appearance when crossing the Alps from Salzburg.

The aim, the designer had in mind, was to obtain high speed and maneuverability.

The cantilever high-wing is in two sections and is trapezoidal; the profiles from root to tip are: NACA 2318, 2315 and 4312. Considerable security against droop has been accomplished by root fairing. This plane can easily be kept in a straight direction by the use of the rudder, even if the elevator is "pulled." The ailerons are rather large and are made of dural; they have "levelling or compensation" tabs.

The fuselage is pulled up and backward, an idea which has proved itself with the "Fledermaus," particularly in bad terrain. Cantilever empennage, both rudder and stabilizer unbraced.

[Endorsed]: Filed August 27, 1951.

[Title of Court of Appeals and Cause.]

STIPULATION RE APPEAL RECORD

It Is Hereby Stipulated by and between the parties to the above-entitled appeal, through their respective attorneys, that the following exhibits and portions of exhibits originally designated for printing but omitted by the printer shall be printed in a supplement to the printed record on appeal.

Defendants' Exhibit A (Report on Airfoil Selection for the Revised Two-Engine Tailless Design ZA-101), pages 1 to 60, inclusive;

Defendants' Exhibit EE (Glen L. Martin Co. Engineering Report No. 1326);

Defendants' Exhibit OOO, last two (2) pages only;

It Is Further Stipulated that the following exhibits, previously designated for printing, need not be printed but may be considered by the Court in their original form without the necessity of reproduction:

Plaintiffs' Exhibit 35;
Defendants Exhibit LL;
Defendants' Exhibit PPP;
Defendants' Exhibit XXX.

Dated September 11, 1951.

HARRIS, KIECH, FOSTER & HARRIS,

By /s/ FORD HARRIS, JR.,
Attorneys for Apellants.

LYON & LYON,

By /s/ FREDERICK W. LYON,
Attorneys for Appellees.

Approved and It Is So Ordered.

........................,
United States Circuit Judge.

[Endorsed]: Filed September 13, 1951.